LEM AND SPACE SUIT ASSEMBLY CONTINGENCY ANALYSIS

APOLLO MISSION PLANNING TASK FORCE

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J. M. Cook R. J. Munz
Prepared by
J. M. Cook R. J. Munz

T. Barnes
Checked by
T. Barnes, AMPTF Chairman

A. B. Whitaker
Approved by
A. B. Whitaker, LEM Systems
Project Engineer

E. Stern
Approved by
E. Stern, LEM Systems
Analysis and Integration

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Assembled by

Grumman Aircraft Engineering Corporation

Bethpage, N. Y.

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Grumman Aircraft Engineering Corporation • Hamilton Standard • Massachusetts Institute of Technology

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SECTION I

PREFACE

I.0

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The following personnel were the principal contributors to this document:

Grumman Aircraft Engineering Corporation

Propulsion	D. Bradley
Reaction Control	R. Pearce, D. Treffeisen
Guidance and Navigation	M. Williams, R. Herrmann, L. Van Dercreek
Stabilization and Control	J. Marino, H. Otten, E. Keating
Electrical Power/Power Generation	H. Bell, A. Gartenberg
Electrical Power/Power Distribution	C. Keenan
Environmental Control	E. Cytryn
Structural/Mechanical	J. Valentine
Communications	V. Silzer
Instrumentation	A. Kromholz, D. Lee
Crew Provisions	H. Churchill
Space Suit Assembly	H. Churchill

Hamilton Standard

Space Suit Assembly	E. Rosenthal

Massachusetts Institute of Technology

Guidance and Navigation	H. F. Little, M. Richter

SUMMARY

The major objective of this contingency analysis, performed under the direction of the Apollo Mission Planning Task Force, was to determine the ability of the Apollo crew to continue their assigned mission or to abort safely when failures occur in any of the Apollo spacecraft subsystems. Subsidiary objectives included identification of situations requiring maximum performance, for use in critical design requirements studies, and assembly of general contingency data for such project use as reliability studies and crew training.

The analysis was performed by systems engineers and subsystem specialists assigned to the AMPTF by NAA, MIT, Hamilton Standard, and GAEC. This report contains the LEM and space suit assembly portions of the analysis. The CSM analysis has been reported separately by NAA.

The latest approved subsystem configuration and the Reference Mission presented in the Phase I Progress Report (Reference 1) were utilized as basic inputs. To complete the study within the AMPTF Phase II effort, only single failures were examined. The study included an analysis of whether the mission could be continued for the particular failures, but no decision was made as to whether it should be continued or aborted.

The results of the analysis indicate that a design philosophy stating that single failures shall not cause loss of the crew has been generally adopted by the LEM project. The few single failures which would cause crew loss have been identified in Table 1 of the Discussion and subsystem redesign to eliminate several of these critical areas, is afoot.

In contrast with the few remaining failures which could cause crew loss, many contingencies would require abort of the mission, and for still more (probably the majority of all possible contingencies) mission abort would be strongly suggested.

When future contingency analysis work is performed, it is recommended that:

- A logical and consistent abort philosophy be determined.
- Periodic updating of this document be performed to include the effect of subsystem configuration and mission plan changes.
- The Phase II Design Reference Mission (Reference 2) be substituted for the Phase I Reference Mission.
- The analysis be expanded to include the effect of multiple independent failures, repair capability, crew incapacity or error, and trajectory effects resulting from propulsion and guidance error.

INTRODUCTION

STUDY OBJECTIVES

One of the objectives of the Apollo Mission Planning Task Force (AMPTF) is to establish mission-related spacecraft functional requirements and to determine the present Apollo spacecraft capabilities relative to these requirements for both nominal and contingency situations. The contingency analysis described in this report was organized to systematically determine Apollo spacecraft capabilities when contingencies have occurred.

The specific objectives of this study were:

- To determine the effect of single failures during the mission on the ability to continue the mission or to abort safely.
- To highlight contingency situations which require greater than nominal performance from the subsystems, for use in the AMPTF study of critical design requirements.
- To provide data which can be used for reliability analysis, crew training, attitude time histories during abort, etc.

STUDY APPROACH

The early phases of the contingency analysis study were concerned with precise definitions of the task to be performed. This was to insure that the study produce results in a form simultaneously useful to both the immediate AMPTF effort and the continuing project efforts. The resulting plan for the analysis was presented in the AMPTF Phase I Progress Report. The details of that plan will not be repeated in this

report. The time which has elapsed since issuance of the Phase I report has not indicated any need to change the approach.

Specialists from the various LEM subsystems were temporarily assigned to the AMPTF to work on contingency analysis and critical design requirements studies. Support in the area of the CSM, Space Suit Assembly, and LEM Primary Guidance and Navigation Subsystem was provided by North American Aviation, Hamilton Standard, and the Massachusetts Institute of Technology. The data contributed by Hamilton Standard and MIT have been incorporated into the respective subsystem sections. The NAA analysis of the CSM was published as NAA report SID 64-1257 (Reference 3).

In order to complete a contingency analysis for a project as complex as Apollo within the timing of the AMPTF Phase I and II efforts, the scope and level of detail of the analysis had to be severely restricted. The areas not considered included crew loading, crew error, alternate missions, failure detection, repair, and multiple independent failures.

One change from the plan presented in the Phase I report was necessary. When the AMPTF was initially organized, one of the first required decisions involved definitions of the extent to which reliability analysis would be performed. It was decided that reliability considerations would enter only into tradeoff studies and that the probability of mission success or crew safety was not to be determined. Abort decisions and feasibility factors have therefore not been included in the analysis because of the substantial effort required. Ground rules, simulation results, failure detectability studies and even estimates of booster reliability would have to be integrated if other than arbitrary abort decisions were to be made, and the AMPTF schedule did not permit this.

The MISSION COMPLETION MODE column of the worksheets thus document methods by which the mission could be completed, not whether it should be. It was impossible, however, to be completely analytic in defining whether continuation was possible. Manual modes of backup operation and failure-by-degree were especially difficult to handle, generally requiring simulation results which were not available. A rule-of-thumb was to indicate a mission completion mode only if the crew has a better than 50% probability of accomplishing the mission objectives. It should be emphasized again that indicating a method of completing the mission when a failure has occurred does not indicate that mission completion is in any way recommended.

In order to determine quantitatively the effect of equipment failure or loss of consumables, a specific mission had to be defined. The Reference Mission issued in the Phase I Progress Report was selected. The only variation of the Reference Mission which was used was the concept of a "shortened" or "minimum" mission. Many of the assumed failures made a lunar stay of 24 hours impossible. If the Reference Mission lunar stay time were strictly adhered to, the MISSION COMPLETION MODE would have read "None" in each case. It was felt that since a shortened mission was certainly an acceptable alternative to the nominal mission, the report would be more informative if means of completing a shortened mission were described.

One of the greatest difficulties in performing the analysis concerned definition of a subsystem configuration. Almost every subsystem either had recently changed, was changing, or was about to change to some degree. If only firm approved configurations were utilized in the analysis, imminent subsystem changes would not be included, and the analysis would be hopelessly outdated on the day it was published. For that reason, judgement was used to predict the subsystem configuration as of Sept. 30, and the

analysis was performed on that subsystem. In most cases the guesses were accurate. In some cases, however, the configuration and analysis had to be updated to make them correct as of the end of September.

DISCUSSION OF RESULTS

Conclusions pertinent to a specific subsystem are discussed in the summary which accompanies each subsystem's worksheets. Reviewing these summaries, it is apparent that in its present state of development, the LEM design enables the crew to return safely to the CSM in the event of almost any single failure. Justification for this degree of crew safety, with its accompanying weight penalty, can be found in the following exchange of views between MSC and GAEC last fall.

In September 1963, GAEC memorandum LMO-540-142 (Reference 4) proposed that a backup not be provided for every component whose failure would affect crew safety, but that reliability of that component should be the sole criterion. In reply, MSC letter SLE-23-63-487 (Reference 5) stated that this concept was unacceptable because of the difficulty in accurately estimating subsystem reliability. The letter further stated that "It is GAEC's responsibility to identify all single failures that could result in loss of the crew and to obtain approval from the NASA before hardware implementation of that design".

This identification of single failures has now been performed and is presented as Table I. The list appears to be rather small, relative to the total number of possible contingencies considered. It was evident during the analysis that subsystem design personnel had already given much attention to this problem and were actively engaged in changing the configuration to reduce this list still further. The draft version of Table 1 contained more critical items, but they were individually deleted when changes were made in the subsystem configuration.

Of the single-failure contingencies resulting in crew loss which now exist, solutions are being studied for several of them. One probable change would include provisions for emergency heating of the fuel

cell cryogenics when the primary water-glycol loop fails, redundant ATCA AC power supplies, and a command receiver to remotely turn on the CSM radar transponder. The remaining failures (such as landing gear failure upon touchdown) are those for which no design change is presently planned because of the anticipated high reliability of these devices.

Tables 2 and 3 are additional examples of data which can be readily compiled by scanning the worksheets. Table 2 lists contingencies which require extra-vehicular transfer capability for crew safety; Table 3 describes the need for CSM rescue. The contingencies listed in both tables would be added to Table 1 (single failures which cause crew loss) if extra-vehicular transfer and CSM rescue were not available.

Tables could also be generated for failures which require attitude changes (to determine RCS fuel penalty, thermal effects, communication problems, etc.), and failures which require instant launch, to name but a few.

One result of the analysis was the training of LEM subsystem specialists in contingency concepts, especially the effect of failures in one subsystem on other subsystems. This training permitted more rapid and thorough completion of the LEM critical design requirements studies by the same personnel.

TABLE 1

SINGLE FAILURE CONTINGENCIES WHICH COULD RESULT IN LEM CREW LOSS*

- Loss of more than 680 ft/sec of ΔV from ascent propulsion after approximately 245 sec of powered descent - may be caused by fuel, oxidizer, or helium leakage, off-mixture ratio burning or low Isp.
- Low thrust or premature cutoff of ascent engine before approximately 385 sec of powered ascent.
- Failure of Power Generation Section hydrogen or oxygen disconnects at lunar launch.
- Loss of three fuel cells by contamination or by loss of primary cooling loop during almost any manned phase.
- Failure of landing gear at touchdown - may cause LEM tipover or preclude ascent.
- Incapacitation of CSM crewman, which results in inability to turn on transponder or rendezvous radar (actually a crew failure).
- Loss of the ATCA AC power supply within the Stabilization and Control Subsystem prior to the completion of powered ascent.
- Massive leak in a non-isolatable area of the Pressure Garment Assembly either on surface or in cabin during lunar exploration phase.

*CSM rescue (rendezvous and/or docking), lunar surface "buddy" rescue, and extravehicular transfer are assumed to be available; explosive failures were not considered.

TABLE 2

SINGLE FAILURE CONTINGENCIES REQUIRING EXTRAVEHICULAR TRANSFER FOR CREW SAFETY

- Loss of the ATCA AC power supply within the Stabilization and Control Subsystem after completion of powered ascent.
- Loss of three fuel cells by contamination or pressure transient after completion of powered ascent.
- Probe and drogue don't engage during rendezvous docking.
- Upper LEM hatch can't be opened after rendezvous docking.
- Drogue can't be removed from tunnel after rendezvous docking.

TABLE 3

SINGLE FAILURE CONTINGENCIES REQUIRING CSM RESCUE
(RENDEZVOUS AND/OR DOCKING) FOR CREW SAFETY

- Loss of 100-680 ft/sec of ΔV from ascent propulsion after approximately 245 sec of powered descent - may be caused by fuel, oxidizer, or helium leakage, off-mixture ratio burning, or low Isp,
- Cutoff of ascent engine between 385-415 sec of powered ascent.
- Loss of +X-axis thruster or thruster command after touchdown.
- Loss of propellant and thrusters from one RCS system after touchdown due to manifold rupture.
- Loss of three fuel cells by contamination or pressure transient after powered ascent.

RECOMMENDATIONS FOR FURTHER STUDY

As indicated in the Study Approach section of the Introduction, the scope and level of the analysis were deliberately limited to permit release of this report before the end of the AMPTF Phase II effort. These limitations should be removed by future analysis. Some recommendations are discussed below. The best order in which to perform them would be in response to current project needs.

- Because of the significant effect that possible failures have on the design and conduct of the mission and on the design of the subsystems themselves, this analysis should be periodically updated. Changes in mission plan, system configurations, and tank sizes should all be considered, probably every six months.

- The Phase I Reference Mission was primarily intended to provide a detailed description of the mission sequence-of-events, and performance aspects were therefore of secondary importance. Thus, when contingencies occurred, there were generally sufficient expendables remaining to perform at least a shortened mission. Preliminary analysis indicates that the results shown in the worksheets would also be applicable to the Design Reference Mission.

- Recommendations for continuing or aborting the mission upon occurrence of a contingency have not been included in this report because they were felt to be outside of the scope of the AMPTF Phase II effort. With respect to the Apollo project, however, a definite need for these recommendations does exist. A general logic for aborting should be adopted to expedite the decision-making process. Samples of GAEC/MSC thinking appear to favor an abort whenever the next failure would cause loss of the crew. It should be realized that, if this view were adopted, achievement of the mission success reliability goals would be extremely difficult. A less pessimistic approach, but one which requires more analytical effort, would be to abort whenever the probability of crew safety dropped below some predetermined level. It is suggested that one of the first uses of this document be in this important area.

- In combination with the above recommendation, the analysis should be expanded to include the effect of multiple independent failures. In order to keep the effort of manageable size, emphasis should first be placed on failures which cause crew loss, followed by those which prevent completion of the mission.

- Future analysis should also consider means for detecting each of the postulated failures, the capability for repair, the effect of crew incapacity or error, the precise trajectory effects of errors in engine on-off commands, etc.

REFERENCES

1) Phase I Progress Report, Apollo Mission Planning Task Force, GAEC Report No. LED-540-7, 4 May 1964 (three volumes), CONFIDENTIAL.

2) Design Reference Mission, Apollo Mission Planning Task Force, GAEC Report No. LED-540-12, 30 Oct 1964 (three volumes), CONFIDENTIAL.

3) Command and Service Module Contingency Analysis Phase I Report, NAA Report No. SID 64-1257, June 15, 1964.

4) Cook, J. M., Single Failure Criterion, GAEC Memorandum No. LMO-540-142, 23 September, 1963.

5) Single Failure Criterion, NASA letter to Grumman, SLE-23-63-487, signed, Joseph F. Shea.

SECTION 1

PROPULSION SUBSYSTEM

1.1 SUMMARY

The analysis indicates that most contingencies can be handled by the subsystem's redundant components. They are, however, four contingencies resulting in mission failure and three resulting in loss of the crew. The mission failure contingencies are:

- Loss of over 750 ft/sec ΔV in descent stage:

  LEM cannot reach the lunar surface safely.
  Possible causes: leakage - loss of 8.8% useable fuel, or loss of 10.1% useable oxidizer, or loss of 6.5% useable helium (blowdown not considered); off-nominal mixture ratio - under 1.41 or over 1.97; a decrease of 10.1% in I_{sp}. None of these losses fall within expected range of variation. The leakage quantities should be detectable. The critical mixture ratios and Isp represent considerable subsystem degradation. It would take a major malfuction to change performance as indicated.

- Descent engine thrust less than approximately 90% of nominal during high thrust phase of powered descent:

  Guidance subsystem cannot handle this thrust error.
  Possible causes: any failure affecting both (redundant) throttle valves (Rocketdyne), or the main throttle linkage (STL), any failure causing line blockage in propellant or helium lines, or combustion instability in the high thrust range. Combustion instability in the high thrust range is unlikely.

- Inability to throttle descent engine below 24.1% of nominal:
  LEM cannot hover or touchdown assuming Design Reference Mission landing weight,
  Possible causes: same as contingency above. Combustion instability in the low range is one of the most prevalent difficulties in deep-throttled engines. For this reason (among others) two throttling techniques are being developed in parallel descent engine programs. The final selection will depend greatly on demonstrating reliable combustion stability in the low thrust range.

- Thrust vector more than 1.6^{o} from C. G. :

  The RCS can compensate for 1.6^{o}. Beyond that the vehicle would tumble.
  Possible causes: catastrophic failure of ascent propellant tanks, or failures affecting gimbal actuators.

The loss-of-crew contingencies are:

- Loss of over 680 ft/sec ΔV in the ascent stage after 245 seconds of powered descent:

  LEM cannot reach 50,000 feet circular orbit even using all RCS propellants available after a nominal descent.
  Possible causes: leakage - loss of 8.5% useable fuel, or loss of 10% useable oxidizer, or loss of 5.1% useable helium (full thrust criterion); mixture ratio under 1.42 or over 1.97; Isp decrease of 10.2%. As in the descent stage these numbers represent significant changes requiring a major malfunction. However, these margins are not greater than the descent stage and perhaps they should be, because loss-of-the crew is more serious than mission failure.

- Ascent engine thrust fails to rise to 50% of nominal shortly after start:

  Ascent engine nozzle may fail by overheating or may be damaged by shock waves reflected from the descent stage if separation does not take place shortly after ignition. A good time estimate is not available (it is probably less than 10 seconds).
  Possible causes: line blockage, propellant or helium. This contingency is not very likely.

- Premature cutoff before approximately 385 seconds of ascent:

  RCS thrusters using cross feed to ascent tanks can enable LEM to achieve a 50,000 foot orbit for CSM rescue after 385 seconds but not before.
  Possible causes for premature cutoff: electrical malfunction causing cutoff signal, combustion instability or other chamber damage. Premature shutdown in the last several seconds of firing a liquid engine is not uncommon, but the time margin available (30-35 seconds before planned cutoff) should provide a good margin of safety.

The calculations were based on the ΔV budget and control weights given in LED-490-100 and the Design Reference Mission given in LED-540-12.

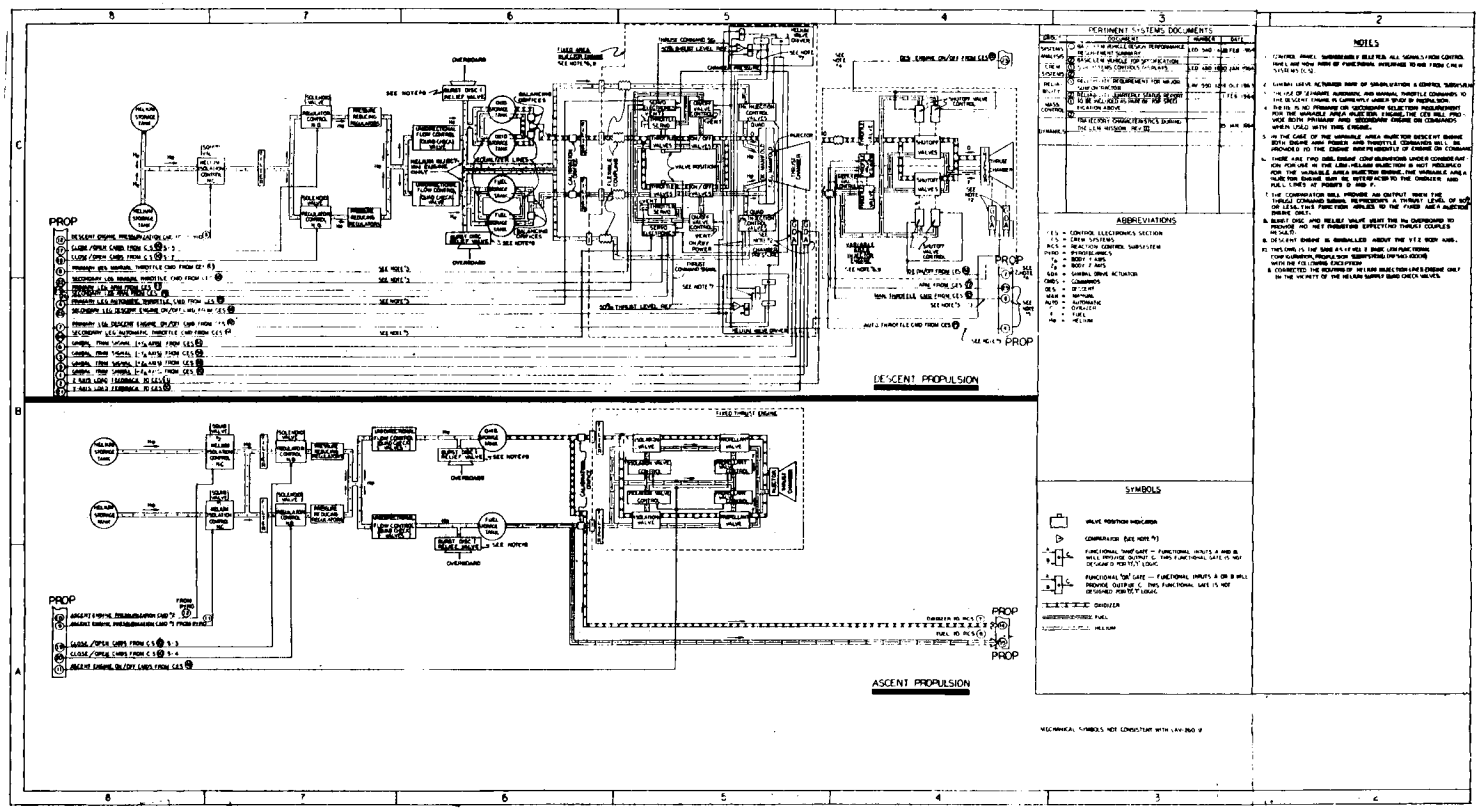

Figure 1-1 Functional Configuration, Propulsion Subsystem

SUBSYSTEM: Propulsion | 1.2 WORKSHEETS | *PHASE:* Earth Launch to Lunar Touchdown

EQUIPMENT/ FUNCTION	*FAILURE MODE*	*UNCOMPENSATED FAILURE EFFECT*	*MISSION COMPLETION MODE*	*MISSION ABORT MODE*	*REMARKS*
Descent Engine/ Available ΔV	Reduced ΔV due to any of the following: • Propellant leakage • Helium leakage • Off-nominal mixture ratio consumption • Reduced Isp	Available ΔV reduced 0-750 ft/sec. Hover and translation capability reduced	Land at primary lunar site. GAEC simulations indicate the 400 ft/sec remaining in the hover and translation allowance suffices to land at the primary landing site.	Abort according to mission time: • Before LEM/CSM separation. Jettison LEM and abort with SPS to primary earth landing site. • During powered descent. Abort with descent engine while propellant lasts. Complete abort with ascent engine. Abort with SPS to primary earth landing site.	Any one of the losses given below will reduce ΔV up to 750 ft/sec. The ΔV budget allows a total of 1150 ft/sec for the phase, hover to touchdown. • Helium 0 - 2.4 lbs (6.5%) • Fuel 0 - 540 lbs (8.8%) • Oxidizer 0 - 990 lbs (10.1%) • Mixture Ratio 1.410 - 1.973 • Isp 0 - 10.1% decrease The percentages quoted show the fraction of useable propellant or helium which is lost. The total propellant tanked was determined by the ΔV budget requirements.
		Cannot land, available ΔV reduced more than 750 ft/sec	None	Same as above.	The ΔV loss could occur as the result of any leakage or off-nominal performance more severe than shown above.
Descent Engine/ Thrust Control	Primary shutoff or primary throttle valves, any malfunction	Thrust error, insufficient thrust, or no throttle	Close primary valves and open alternate valves.	Depends on time of flight. See statemend made for Descent Engine/Available ΔV.	
	Insufficient thrust due to failure affecting both valve sets, or due to line blockage	Maximum available thrust is between 90% and 100% of nominal.	Normal		Propellant requirements and guidance system considerations are met.
		Maximum available thrust is 50 - 90% of nominal	None		Preliminary estimates indicate guidance system cannot handle thrust.
		Thrust below 50%	None		This is listed to show the limit due to available propellant.
	Cannot throttle down all the way. Failure affects both valve sets	Throttle below 24.1% of nominal	Normal to complete the design reference mission.		For the control weight LEM this number is 21.8%. (Based on Table 4 of Report LED-490-100 dated 30 April 1964)
		Throttle down between 24.1% and 25.1% of nominal	Use RCS 4 Thruster Logic in-X axis translation to complete design reference mission.		For the control weight LEM this range is 21.8% to 22.7%.
		Cannot reach 25.1%	Cannot complete design reference mission.		For the control weight LEM this number is 22.7% of nominal.
Descent Engine/ Regulated Pressure	Oscillates	Actuate relief valve	Close solenoid in leg with bad regulator and continue with redundant regulation.		
Descent Engine/ Thrust Alignment	C.g. greater than 6^{o} away from x axis (e.g., loss of Ascent Propellant Tank see RCS)	Engine gimbal limit is exceeded, vehicle tumbles	RCS plus gimbal can maintain stability for misalignment up to 7.6^{o}. Over 7.6^{o}, no mission completion mode is possible.		Approximate RCS propellant consumption penalty - 100 lbs per degree of imbalance. With gimbal hard over RCS can maintain stability up to 7.6^{o}. Beyond this the descent engine must be throttled (see Insufficient Thrust Contingency above).

SUBSYSTEM: Propulsion

PHASE: All LEM Phases

EQUIPMENT/ FUNCTION	*FAILURE MODE*	*UNCOMPENSATED FAILURE EFFECT*	*MISSION COMPLETION MODE*	*MISSION ABORT MODE*	*REMARKS*
Ascent Engine, Available ΔV	Propellant leakage, or helium leakage, or off-nominal mixture ratio burning, or decreased Isp	Stage ΔV available decreases 0-100 ft/sec	Normal (moment control by RCS unbalanced couple mode).	Abort mode depends on time and mission phase. During use of ascent engine, abort and continue modes are identical. For other phases see the first page of this section.	Throughout these calculations the RCS is assumed to have 406 lbs of propellant prior to lunar liftoff. During powered ascent there is assumed to be a 500 ft lb error torque which is controlled by the RCS in the unbalanced couple mode. The decrease of 100 ft/sec does not affect powered ascent for a single failure because this is the allowance for use of AGS. The losses which correspond to this ΔV reduction are: • Fuel 0 - 20 (1%) • Oxidizer 0 - 40 (1.3%) • Helium 0 - 0.05 (.4%) • Mixture ratio 1.59 - 1.63 • Isp 0 - 1.6%
		Stage ΔV available decreases 100-567 ft/sec	Complete injection using RCS propellant for needed ΔV. CSM rescue needed.	Same as above.	This maneuver requires guidance system accomodation of the low thrust-to-weight ratio of RCS thrusters. CSM must perform midcourse corrections, rendezvous and docking, but descent to 50K orbit is not needed. The losses are: • Fuel 20 - 130 lbs (1 - 7 %) • Oxidizer 40 - 250 lbs (1.3 - 8%) • Helium 0.05 - 0.5 lbs (.4 - 4.3%) • Mixture ratio 1.45 - 1.59, 1.63 - 1.90 • Isp 1.6 - 8.9%
		Stage ΔV available decreases 567-680 ft/sec	Complete powered ascent using RCS propellant for needed ΔV. CSM rescue is required. LEM is in a circular orbit with 50,000 foot clear pericynthion.	Same as above.	As above, the guidance system must handle the low T/W. The CSM must descend to 50K to rescue LEM. The losses are: • Fuel 130 - 160 lbs (7 - 8.5%) • Oxidizer 250 - 300 lbs (8 - 10%) • Helium 0.5 - 0.6 lbs (4.3 - 5.1%) • Mixture ratio 1.415 - 1.450, 1.900 - 1.965 • Isp 8.9 - 10.2%
		Stage ΔV available decreases more than 680 ft/sec	None	None	The losses are: • Fuel over 160 lbs (8.5%) • Oxidizer over 300 lbs (10%) • Helium over 0.6 lbs (5.1%) • Mixture ratio under 1.415, over 1.965 • Isp over 10.2% * Wherever helium loss is considered, the reference is to mission completion at full thrust. Maximum helium loss equals one bottle. This leaves one bottle, which is enough to complete an abort ascent at reduced ullage pressure-reduced thrust (blowdown mode).

SUBSYSTEM: Propulsion

PHASE: All After Lunar Touchdown

EQUIPMENT/ FUNCTION	*FAILURE MODE*	*UNCOMPENSATED FAILURE EFFECT*	*MISSION COMPLETION MODE*	*MISSION ABORT MODE*	*REMARKS*
Ascent engine	Rises slowly at ignition	Nozzle damaged by overheating or by reflected shock waves	None.	None.	
	Does not rise above 50%	No liftoff	None.	None.	Thrust-to-weight ratio insufficient for liftoff. By the time propellant consumption reduces weight significantly, nozzle would be overheated.
	Terminates prematurely	Shut down before 385 sec	None.	None.	LEM is below 35,000 - ft clear pericynthion.
		Shutdown after 385 sec	Fire RCS thrusters using crossover to main propellants.	Same as mission continue mode.	RCS thrust can bring LEM up to 35,000-ft clear pericynthion. CSM rescue is required. The time is approximate.

Propulsion Subsystem (Cont.)

1.3 DESCRIPTION

The LEM has two independent propulsion subsystems (Figure 1-1), a subsystem with variable thrust magnitude and direction, for the descent stage and a subsystem with constant thrust in a fixed direction for the ascent stage. Both propulsion subsystems basically consist of three integrated sections. The first of these is the propellant tank pressurization section which stores and regulates the flow of the high pressure gas required for pressurization of the propellant tanks. The second section is the propellant storage and supply section. The rocket engine section then utilizes the propellant flow to provide the desired thrust magnitude

1.3.1 Descent Propulsion Subsystem

1.3.1.1 Propellant Tank Pressurization Section: As shown in Figure 1-1, the propellant tank pressurization section comprises high pressure helium storage tanks, valves and regulators for the control of the flow of helium, and the associated piping.

The gaseous helium is stored at 3,500 psia pressure in each of the two manifolded storage tanks, and is isolated from the other subsystem sections prior to the first engine ignition by an explosive squib valve at the manifold outlet. Subsequent to the firing of the squib valve, the helium flows downstream through either one of two parallel paths. Each path contains a normally open latching solenoid shut-off valve followed by a pair of series mounted pressure regulators. In the event of failures, either path may be closed-off during engine operation by energizing the solenoid shut-off valve. The pressure settings of the regulators are staggered so that, if the controlling regulator fails open or closed, the downstream regulator or the parallel leg will take over. Because of the difference in the pressure control settings among the four pressure regulators, one path is always closed off during engine operation. This

redundancy provides for normal operation in the event of any regulator failure. After flowing through one of the two parallel flow paths, the pressure regulated helium flow branches into two flow paths - one that leads to the fuel tanks and the other that leads to the oxidizer tanks. Each of these two paths contains a quad check valve to prevent backflow and mixing of propellant vapors, a burst disc and a relief valve to prevent over pressurization of the propellant tanks.

1.3.1.2 Propellant Storage and Supply Section: The descent stage propellant storage and supply section (see Figure 1-1) consists of two fuel tanks, two oxidizer tanks, two filters and piping. Each set of like tanks contains four interconnecting lines - one each for gas and liquid transfer and one each for gas supply and liquid discharge. The two gas lines enable the tanks to be maintained at equal pressures while the two liquid lines enable both like tanks to empty simultaneously, thus minimizing c.g. movement. The propellant from like tanks which is under the action of helium pressure is forced through the manifolded liquid transfer line through a filter and into the rocket engine section.

1.3.1.3 Rocket Engine Section: Currently two rocket engine designs are being developed for the variable thrust descent engine - one by Rocketdyne and the other by STL. Both engines have the capability of varying the thrust magnitude and direction. The main difference between the two is the technique and the hardware with which thrust magnitude is controlled. Both engines are gimballed to provide thrust direction variation.

1.3.1.3.1 Rocketdyne Engine: The Rocketdyne Engine consists of a thrust chamber assembly, a primary propellant valve assembly, a redundant alternate valve assembly, helium control valves, and associated

propellent and helium lines. The primary valve assembly regulates propellant flow to the thrust chamber assembly where the propellants react. The redundant alternate valve assembly is used only if the primary set malfunctions.

Each propellant valve assembly includes: a fuel throttle valve, a fuel shutoff valve, an oxidizer valve, an oxidizer shutoff valve, and a servo control valve. In a normal start the throttle and shutoff valves open fully. To throttle, the throttle valves close partly and the shutoff valves remain wide open. The fuel and oxidizer throttle valves are mechanically linked and only the fuel side is driven by an external actuator. The actuator is controlled by the servo control valve which responds to commands from the Stabilization and Control section. At cutoff the throttle and the shutoff valves close completely.

The thrust chamber assembly is comprised of propellant manifolds and injector, a combustion chamber, a nozzle and nozzle extension. Fuel and oxidizer coming from the primary valve assembly are distributed across the rear face of the injector, in the injector fuel and oxidizer manifolds. Orifices in the injector determine the direction and velocity of propellant streams entering the combustion chamber. The gases produced by the propellant reaction then flow out of the combustion chamber through an ablatively cooled nozzle which is terminated at an area ratio of 6 to 1 and continues through a radiation cooled nozzle extension to an area ratio of 53 to 1. At thrust levels below 50% maximum thrust helium is mixed with the propellants in the injector manifolds in order to maintain injector pressure drops commemsurate with combustion stability. Fuel and oxidizer helium flows are initiated simultaneously.

1.3.1.3.2 STL Engine: The STL engine is similar to the Rocketdyne engine except for the method of controlling propellant flow. The propellant flow control valves in this engine consist of a pair of mechanically linked variable area venturis which in turn are mechanically linked to an element in the variable area injector. Thus, the thrust commands that effect the positioning of the propellant flow control valves automatically cause the positioning of the variable area injector.

The thrust chamber assembly is similar to that of the Rocketdyne Engine except that the nozzle is terminated at an area ratio of 49 to 1.

1.3.2 Ascent Propulsion System

1.3.2.1 Propellant Tank Pressurization Section: This section consists of two independent high pressure helium storage tanks that contain approximately 13.5 lbs. of a helium, explosive squib valves, normally open latching solenoid valves, pressure regulators, quad check valves and relief valves. After being filtered and regulated to the proper pressure the helium flows into the Propellant Tank Storage and Supply Section.

1.3.2.2 Propellant Tank Storage and Supply Section: The Propellant Tank Storage and Supply Section for the ascent stage includes one fuel tank, one oxidizer tank and piping. The tanks, which are loaded in the oxidizer-to-fuel weight ratio of 1.6 to 1.0, are located at unequal distances from the X axis. This arrangement makes their moment contributions equal so that c.g. offset is minimized. The propellant, which is under pressure from the helium within the tanks, is available to both the Rocket Engine Section and the Reaction Control System.

1.3.2.3 Rocket Engine Section: This section basically consists of filters, propellant shut-off valves, valve controls and a thrust chamber assembly. For redundancy, the individual fuel and oxidizer lines that enter the Rocket Engine Section are split into two fuel and two oxidizer lines (a primary and a secondary) each of which contains a filter and two shut-off valves in series. Each oxidizer valve is linked to a like fuel valve. During start-up the engine "on" command is transferred to the shut-off valve controls which in turn causes the opening of all four pairs of shut-off valves simultaneously.

SECTION 2

REACTION CONTROL SUBSYSTEM

2.1 SUMMARY

The contingency analysis for the RCS subsystem indicates two single failure situations which greatly affect the ability of LEM to perform its mission. These failures are:

- Total loss of propellant from either one of the two RCS tankage systems.
- Failure of a manifold which results in total loss of propellant from either one of the two RCS tankage systems and in addition loss of eight thrusters.

Total loss of propellant from either tankage system affects the LEM capability in a manner which is dependent upon mission phase as follows:

- Earth launch to lunar touchdown - mission may be completed with CSM rescue of LEM from 50,000 feet circular orbit. Aborts during LEM powered descent may be made without CSM rescue if the loss occurs early enough in the powered descent so that the remaining RCS propellant plus the unused ascent engine propellants are sufficient for a LEM rendezvous and docking maneuver. Aborts made late in the powered descent will require CSM rescue.
- Lunar touchdown to initiation of rendezvous - LEM return to the CSM is achievable with varying amounts of CSM assistance. If RCS propellant loss occurs early in the powered ascent, the CSM must expend sufficient ΔV to perform the rendezvous and docking maneuvers. If the failure occurs late in the powered ascent or after powered ascent but prior to initiation of rendezvous, the LEM can perform the rendezvous maneuver but will require CSM active docking.
- Initiation of rendezvous to final docking - RCS tankage failure during the final LEM phase does not preclude the ability of LEM to perform both rendezvous and docking.

Failure of one of the two RCS manifolds during any LEM phase precludes a mission completion. Safe aborts with this failure are achievable at any time provided the CSM rescue of LEM is possible. Manifold failure within the RCS results in loss of 50% of the RCS propellant at the time of failure. Consequently, this type of failure is identical to the previously described failure and its failure effects. In addition, the loss of eight thrusters resulting from failure of a manifold reduces the control capability of LEM with the result that LEM may not be able to perform the lunar landing and final docking maneuvers.

The contingency analysis also shows that no single thruster failure results in loss of mission completion or safe abort capability. One interesting single thruster failure involves the +X-axis thrusters. If a +X-axis thruster fails on during powered descent, more propellant may be used during powered ascent if the failed thruster is turned off than if the failure is left uncorrected. Since the nominal LEM ascent uses unbalanced couples to provide simultaneous attitude control and ΔV along the flight path, loss of a +X-axis thruster and subsequent shutoff will require a switch to balanced couple mode for attitude control. Hence, the 213 fps ΔV nominally derived from the RCS operating in unbalanced couple mode would be lost. The ascent engine would therefore be required to make up this ΔV loss by burning for a longer time. The longer burn time would require additional propellant from the main tanks and would therefore use up the potential backup propellant for the RCS.

2.1.3.4

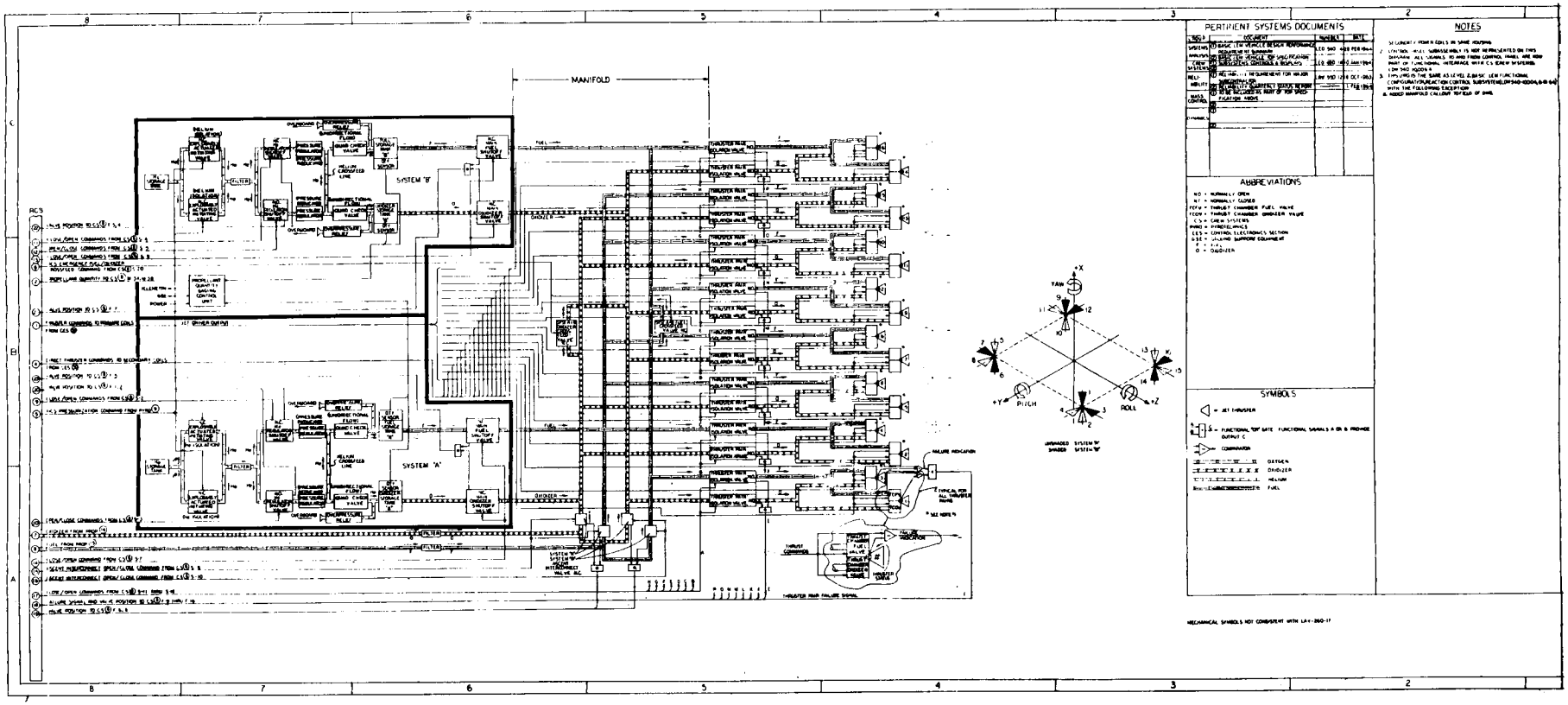

Figure 2-1 Functional Configuration, Reaction Control Subsystem

SUBSYSTEM: Reaction Control

2.2 WORKSHEETS

PHASE: All LEM Phases

EQUIPMENT/ FUNCTION	*FAILURE MODE*	*UNCOMPENSATED FAILURE EFFECT*	*MISSION COMPLETION MODE*	*MISSION ABORT MODE*	*REMARKS*
Any component in He press. system, propellant tankage section, or main s/o valve	Rupture, external leak, or blocked line in areas other than squib valve, regulator s/o valve, press. regulator, or check valves. These areas excluded because parallel flow paths exist.	Loss of propellant from one of two redundant systems	Close N.O. regulator s/o valve and main s/o valves of failed system, and open N.C. crossfeed valves to feed all 16 thrusters from remaining system. • Separation to initiation of rendezvous: None. (1) • Initiation of rendezvous to crew transfer: Normal	Close N.O. regulator s/o valve and main s/o valves of failed system, and open N.C. crossfeed valves to feed all 16 thrusters from remaining system. • Separation to insertion: Dock to CSM. • Insertion to touchdown: Abort with descent and/or ascent engine. Dock to CSM. (2) • Lunar stay to initiation of rendezvous: Abort with ascent engine and have CSM rescue LEM. • Initiation of rendezvous to crew transfer: Continue nominal mission and dock to CSM.	Abort is possible with descent engine alone for 245 sec after powered descent initiation; thereafter, staging is necessary. (1) If failure occurs between separation and touchdown, the mission can be completed only if CSM rescue is provided. If failure occurs after powered ascent and before MCC's, then only CSM dock is necessary. * This assumes that the ascent propellant is used for RCS thrusters during powered ascent. When ascent supply is depleted the RCS propellant & thrusters supplant the ascent engine. (2) Aborts during descent can be completed without CSM rescue only if RCS thrusters make use of excess ascent propellant.
Main s/o valve, isolation valve, crossfeed valve, or ascent interconnect valve	Rupture or external leak on manifold side	Loss of propellant and thrusters from one of two systems	None. (See Remarks.)	For all phases, close N.O. regulator s/o valve and main s/o valve and operate with remaining eight thrusters. • Separation to insertion: CSM docks with LEM. • Insertion to touchdown: Abort with descent and/or ascent engine. CSM docks with LEM. • Lunar stay to initiation of rendezvous: Abort to contingency orbit and have CSM rescue LEM. • Initiation of rendezvous to crew transfer: Continue nominal ascent and have CSM dock to LEM.	This failure precludes a mission completion mode regardless of phase. That is, lack of LEM maneuverability, resulting from the loss of one system's thrusters, makes necessary CSM docking. With one system's thrusters inoperable, normal logic precludes: pure Z-axis translation; translation in one of the Y-axis directions; pure yaw or rotation about X-axis. If failure occurs prior to liftoff, 213 fps from RCS is lost during powered ascent. If failure occurs during powered ascent, some fraction of the 213 fps is lost. This ΔV is provided by ascent engine; therefore, ascent propellant, which could have been used for RCS thrusters during powered ascent, is now unavailable. If this failure occurs anywhere after start of hover then the mission can be completed if CSM rescue is provided. This assumes that the ascent propellant is used for RCS thrusters during powered ascent. When the ascent supply is depleted the RCS propellant and thrusters supplant the ascent engine. Prior to this point there is insufficient propellant to continue to the lunar surface and ascend to a safe rescue altitude.
Pressure regulator in active leg	Fails closed	Loss of propellant from one of two redundant systems	Close N.O. regulator s/o valve and open N.C. regulator s/o valve. Continue nominal mission.	Close N.O. regulator s/o valve and open N.C. regulator s/o valve. • Separation to insertion: Dock to CSM. • Insertion to touchdown: Abort with descent and/or ascent engine. • Lunar stay: Abort with ascent engine. • Liftoff to docking: Continue nominal ascent and dock to CSM.	Failure of a passive pressure regulator in closed position or failure of any regulator in open position has no effect other than loss of some backup capability.
Isolation valve	Fails open	None	Nominal.	Not applicable.	Open position is normal for isolation valve. This failure is significant only when some other contingency calls for closing isolation valve.

SUBSYSTEM: Reaction Control

PHASE: All LEM Phases Except Rendezvous

EQUIPMENT/ FUNCTION	*FAILURE MODE*	*UNCOMPENSATED FAILURE EFFECT*	*MISSION COMPLETION MODE*	*MISSION ABORT MODE*	*REMARKS*
Thrusters 1-16, injector valve, or isolation valve	Thruster: fail on or off Injector valve: fail open, external leak, or rupture. Isolation valve: rupture, external leak on thruster side, or fail closed.	Loss of propellant from one of two redundant systems	Close appropriate isolation valve to shut off thruster pair. Logic automatically selects proper alternate thrusters.	Close appropriate isolation valve to shut off thruster pair. Logic automatically selects proper alternate thrusters. • Separation to insertion: LEM or CSM docks. • Insertion to touchdown: Abort with descent and/or ascent engine. • Lunar stay: Abort with ascent engine. • Liftoff to docking: Continue nominal ascent and dock to CSM.	LEM or CSM will be active in docking depending on specific thruster shut off. If failure results in loss of a +X-axis thruster, c.g. compensation during remaining part of powered ascent is performed with balanced couples. Hence, there is no boost of 213 fps by RCS and the main engine, therefore, burns longer and consumes additional propellant. This precludes the use of that ascent propellant for RCS thrusting during powered ascent.
Injector valve	Fails closed	See Remarks	Close appropriate isolation valve to shut off thruster pair. Logic automatically selects proper alternate thrusters.	Same as mission completion mode.	No thrust when commanded. Hence, rotational maneuvers will require twice normal time. Any maneuver, using this thruster, will induce disturbance torques requiring additional propellant for compensation. LEM or CSM will be active in docking depending on specific thruster shut off. If failure results in loss of a +X-axis thruster, c.g. compensation during remaining part of powered ascent is performed with balanced couples. Hence, there is no boost of 213 fps by RCS and the main engine, therefore, burns longer and consumes additional propellant. This precludes the use of that ascent propellant for RCS thrusting during powered ascent.

SUBSYSTEM: Reaction Control

PHASE: Rendezvous

EQUIPMENT/ FUNCTION	*FAILURE MODE*	*UNCOMPENSATED FAILURE EFFECT*	*MISSION COMPLETION MODE*	*MISSION ABORT MODE*	*REMARKS*
Z-axis thrusters 3 or 15, or injector valves for these thrusters.	Thruster: fail on or off; Injector valve: rupture, external leak, fail closed, or fail open.	Thruster on: Additional propellant consumed from thruster on and from automatic yaw torque compensation. Thruster off or injector valve fails closed: Additional propellant consumed by uncalled-for torque induced by single thruster firing when a pair is commanded. Injector valve external leak, rupture, or fail open: Loss of propellant from one of two systems.	Close appropriate isolation valve to shut off thruster pair. Logic automatically selects proper alternate thrusters. LEM continues phase in automatic mode. Alternatively, manually pitch LEM to align +X-axis with CSM. In manual attitude hold mode, retrofire X-axis thrusters 5 and 13 for all rendezvous burn. Manually control impulse applied by monitoring ΔV display.	Same as mission completion mode.	Nominal rendezvous ΔV required = 198 fps • LOS rate component = 48.5 fps (along X) • Range rate component = 149.5 fps (along Z) For the first continue mode, the operable thruster burns for an additional 127 sec (total of all burns). Yaw torque compensation with thrusters 8 and 16 is automatically commanded in pulse mode. Additional ΔV required for all burns (due to pulsed yaw torque at degraded Isp is 163.6 fps. Total ΔV required increases to 361.6 fps. However, only 293 fps is available from the RCS tankage and only 48.5 fps can be taken from the 83 fps remaining in the ascent tankage (ascent propellant unavailable for Z-axis burns). Hence, there is insufficient ΔV for this rendezvous and CSM docking is required. For the alternate mode, total ΔV for rendezvous is 198 fps, compared with 293 fps available. Above remarks assume that rendezvous is flown in manual mode (with AGS). This contingency implies a double failure situation since a nominal ascent is performed with PNGS. If rendezvous is performed with the PNGS a single thruster failure (X or Z-axis) will not increase ΔV expenditure because the nominal rendezvous burn is with the X-axis thrusters which are totally redundant for single failures.
Any thruster except 3 and 15, injector valve of any thruster except 3 and 15	Same as above	Thruster on: Additional propellant is consumed from thruster on and from automatic compensation of induced torque. Thruster off or injector valve failed closed: Additional propellant consumed by uncalled-for torque induced by single thruster firing when a pair is commanded. Injector valve external leak, rupture, or fail open: Loss of propellant from one of two systems.	Close appropriate isolation valve to shut off thruster pair. Logic automatically selects proper alternate thrusters. Continue nominal mission. (See Remarks.)	Same as mission completion mode.	Docking may require an active CSM depending upon specific thruster failure.

Reaction Control Subsystem (Cont.)

2.3 DESCRIPTION

The Reaction Control Subsystem (RCS) provides the impulses that control the LEM in six degrees of freedom during all of the LEM inflight phases. The RCS consists of two independent but interconnectable bi-propellant sections, two helium pressurization sections and sixteen 100-pound-thrust chambers that are arranged in clusters of four around the periphery of the LEM ascent stage (see Figure 2-1).

The RCS may be operated in either a fully automatic, semi-automatic or manual mode. In the fully automatic mode, RCS control commands originate in the primary navigation and guidance subsystem. Semi-automatic mode operation combines automatic attitude-hold with manual control available on each axis with the astronaut's hand controller. In the manual mode, control commands and duration of RCS thrust originate at the astronaut's hand controller. A pulse modulator and jet logic network process the RCS control commands and establish which of the sixteen thrust chambers is to be fired. Attitude and translational control are provided by appropriate signal routing in the logic section of the stabilization and control system.

2.3.1 Helium Pressurization Section

Propellant (fuel and oxidizer) tanks are pressurized by regulated helium gas acting upon the propellant tank bladders. Gaseous helium is stored in two spherical titanium tanks at 3,000 psi and 70°F. Two redundant explosively operated squib valves seal each helium tank until just prior to LEM separation from the CSM. A filter downstream of the squib valves traps possible debris and contaminants.

Downstream from the filter, each helium pressurization section splits into redundant lines. One line begins with a normally open, solenoid-operated, latch-type shut-off valve while the other line begins with a normally closed shutoff valve. Only one helium line is open at a time. Should this line fail, the redundant line is opened. A two-stage line sensing pressure regulator follows each shutoff valve. Each regulator is capable of reducing the helium pressure to propellant operating pressure; however, only one regulator in the line functions while the second remains passive. The single open helium line then splits into an oxidizer tank line and a fuel tank line. Quad check valves, one on each propellant tank line, prevent backflow and ensure isolation of the pressurizing helium and of one propellant tank from the other.

A relief valve, close to each helium port on the propellant tanks, is set to relieve at 250 psi.

2.3.2 Propellant Sections

Each of the two propellant sections contains two tanks, located in the LEM ascent stage. Each propellant tank contains a 3-ply Teflon bladder supported by a standpipe running lengthwise in the tank. The oxidizer tanks are loaded with nitrogen tetroxide (N_2O_4) and the fuel tanks are loaded with a 50-50 mixture of hydrazine (N_2H_4) and unsymmetrical dimethylhydrazine (UDMH). Although the tanks are sized for 614.4 lbs. of propellant (203.7 lbs. N_2O_4 and 103.5 lbs. UDMH per system), they are presently loaded to 549.6 lbs. total, or 274.8 lbs. per system. Radiation-type quantity gauging devices, using sensors external to the propellant tanks, enable the quantity of propellant to be measured.

Latch-type, solenoid-operated shutoff valves permit isolation of any pair of four propellant tanks. From this valve the propellant flows into a manifold feeding the eight thrusters. The manifolds from the two independent propellant sections can be connected through the actuation of a normally closed solenoid valve.

Capability of feeding the RCS thrusters from the ascent propellant tanks is incorporated with four normally closed, solenoid-operated, latch-type shutoff valves.

Sixteen isolation valves are provided to allow for manual cutoff of propellant flow to adjacent thruster pairs.

2.3.3 Thrust Chamber Assembly

The thrust chamber assemblies section consists of sixteen thrusters grouped in clusters of four. Each thruster is capable of operating in a pulse or steady state mode. The propellant lines, valves and injectors are encased in a radiation shield and insulation blanket that provides protection from solar radiation and radiation from the operating engine while at the same time radiating thrust-chamber-generated heat. There are two injector valves for each thruster. One introduces oxidizer to the thrust chamber while the second introduces fuel. They are double coiled, electrically operated valves which receive their command signals from the control electronics section.

SECTION 3

PRIMARY GUIDANCE AND NAVIGATION SUBSYSTEM

3.1 SUMMARY

The contingency analysis for the Guidance and Navigation Subsystem indicates only one single failure contingency which may result in loss of the crew. The contingency occurs if the CSM crewman is disabled at a time in the lunar orbit when the CSM rendezvous radar transponder is normally off. Inability of the CSM crewman to turn on the transponder results in functional loss of the LEM RR. The significance of this loss requires that ways be studied to turn on the CSM transponder by remote control from LEM or GOSS. A possible solution might be to incorporate a turn "on" via a command receiver operating on VHF.

Single failures occurring within the IMU, CDU, PSA, LR, or LGC prior to hover in the powered descent will, in general, not permit a safe landing. However, safe aborts can be achieved during either powered descent or ascent with the Abort Guidance Section.

Loss of the LEM RR prior to powered descent prevents a mission completion. Safe abort, however, is possible using the CSM RR, LEM transponder and the VHF voice link to effect mid-course corrections and rendezvous maneuvers.

Loss of the LEM Landing Radar (LR) prior to the hover point prevents mission completion. A preliminary calculation indicates that a safe landing is possible without the LR provided the loss of the LR does not occur before LEM reaches an altitude of approximately 2000 feet. At about this altitude, the LGC will have been updated sufficiently to allow a completion of the powered descent using inertial information only. Further study is recommended to substantiate this preliminary result.

Present design of the LR includes a two-position antenna; position one is for use prior to hover and position two is for use from hover to touchdown. The value of LR data derived during the hover to touchdown phase is uncertain due to the high probability that lunar dust and engine plume effects may disrupt normal LR operation. It is presently assumed that the inertial reference within the IMU will be sufficient to allow normal descent from hover to touchdown. It is therefore recommended that further study be undertaken to determine if the LR antenna can be simplified to a fixed-position configuration.

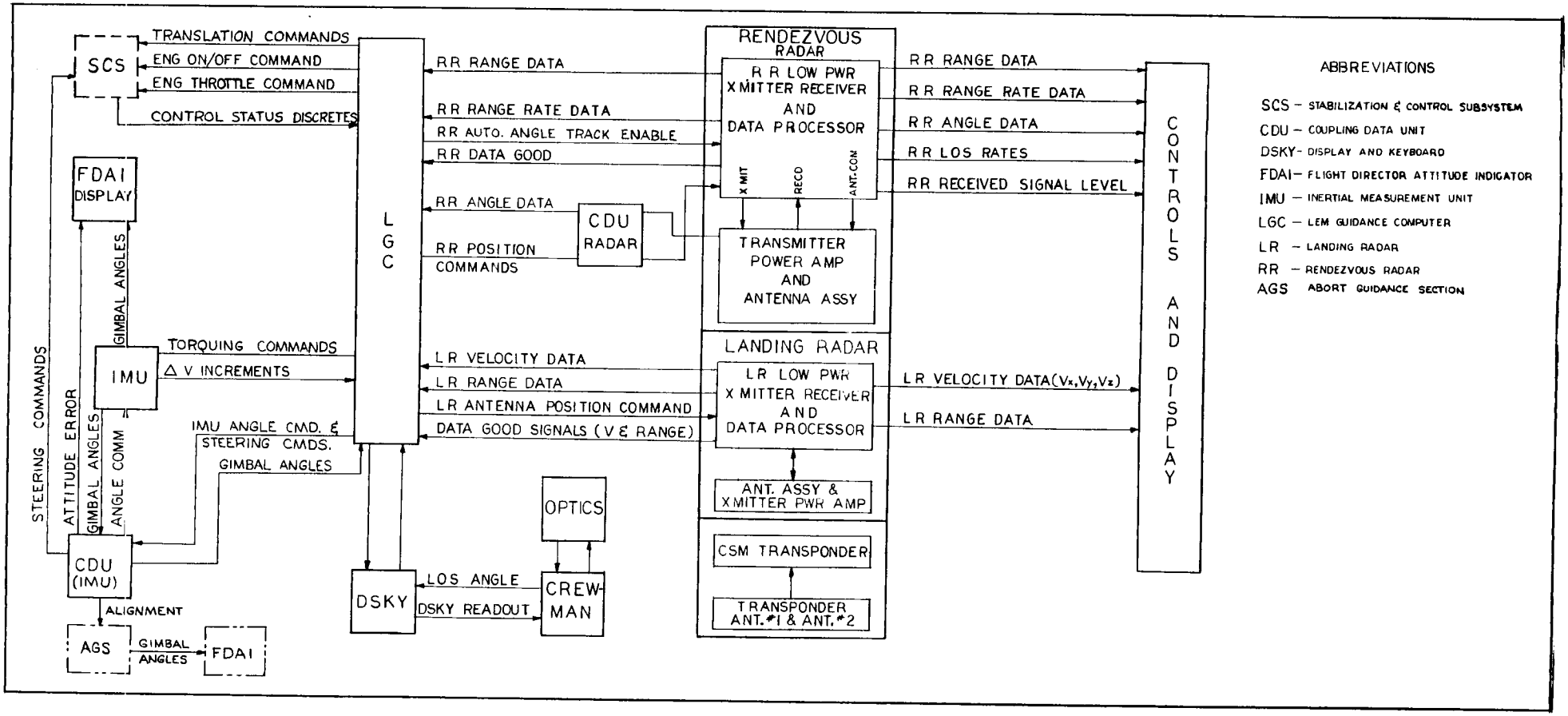

Figure 3-1 Functional Configuration, Guidance and Navigation Subsystem

SUBSYSTEM: Primary Guidance and Navigation 3.2 WORKSHEETS *PHASE:* All LEM Phases

EQUIPMENT/ FUNCTION	*FAILURE MODE*	*UNCOMPENSATED FAILURE EFFECT*	*MISSION COMPLETION MODE*	*MISSION ABORT MODE*	*REMARKS*
IMU, total attitude	Erroneous CDU outputs	Incorrect thrusting	Prior to hover: none. Hover to touchdown: proceed with manual control. Any phase after touchdown: switch to AGS and manually orient LEM for midcourse corrections.	Abort to CSM using AGS.	
	Loss of inertial reference	Incorrect thrusting and loss of attitude presentation on primary FDAI			
IMU, ΔV increment	Erroneous or no output	Incorrect thrusting	Prior to hover: none. Hover to final docking: proceed under manual control.		
LGC/CDU, attitude error display signal	Incorrect FDAI display	Loss of FDAI display readout	Switch AGS FDAI to PGNS.		
LGC/CDU, RR position commands	Erroneous CDU outputs	Radar follows incorrect slewing commands	Prior to hover: none.(1) Hover to final docking: operate RR manually or use CM RR data and input via DSKY.	Use PGNS and CM RR data.	(1) If failure occurs near end of powered descent (hover), mission completion is possible.
LGC, IMU commands	No output	Cannot realign IMU	Prior to touchdown: proceed without alignment update (1). Lunar surface operations prior to liftoff: align AGS and perform ascent.	Prior to touchdown: use PGNS After touchdown: use AGS	(1) Simulation studies will determine if ΔV penalties which result are prohibitive.
LGC, LR position commands	No output	LR is not properly positioned prior to hover, therefore cannot land	Prior to hover: none.(1) Hover to touchdown: land using inertial displays.	Return to CSM using PGNS.	(1) Loss of LR prior to approximately 2000 ft altitude will not permit a landing. Studies are being performed to determine if the RR may be used as a backup to the LR.
LGC displays	Lights inoperative	Loss of failure detection capability	Proceed with risk of undetectable subsequent failures.	Same as above.	
	DSKY inoperative	Cannot use DSKY to display parameters	Prior to touchdown: none.		
LGC steering commands	No commands or erroneous commands	Incorrect thrust vectoring or incorrect attitude control	Prior to hover: none. Hover to touchdown: continue under manual control. After touchdown: switch to AGS.	Return to CSM using AGS.	Loss of control about a single axis may be compensated for by the computer with a ΔV penalty.
LGC, engine thrust commands	No commands or erroneous commands	Incorrect thrusting	Prior to start of powered descent: none After start of powered descent: continue with PGNS using manual control.	Same as above.	
LGC, Sync signals	Loss of one or more strobe or sync signals	Loss of information from RR to LGC and from LR to LGC	Prior to hover: none.(1) Hover to touchdown: switch to PGNS inertial displays. After touchdown: use CM RR data derived via VHF voice link and input to LGC via DSKY. LEM RR data may be monitored and compared with CM RR data.	Abort using PGNS and CM RR information.	(1) If failure occurs near end of powered descent (hover), mission completion is possible.
Optics, LOS angle measurements	Information not entered (mark fail)	Unable to complete alignment	Prior to powered descent: proceed without alignment. Update and suffer ΔV penalty and/or landing site error. (1)	Return to CSM using PGNS.	(1) Simulation studies will determine if ΔV penalties which result are prohibitive.
RR automatic angle track enable	No output	RR will not operate in automatic track mode	Prior to powered descent: none. All other phases: Operate RR manually or use CM RR data and input via DSKY.	Use PGNS and CM RR data.	

SUBSYSTEM: Primary Guidance and Navigation

PHASE: All LEM Phases

EQUIPMENT/ FUNCTION	*FAILURE MODE*	*UNCOMPENSATED FAILURE EFFECT*	*MISSION COMPLETION MODE*	*MISSION ABORT MODE*	*REMARKS*
Landing Radar (LR) velocity data (digital only)	Velocity data (one or all) cannot be read into LGC	LGC cannot improve (update) its knowledge of velocity	Prior to hover: none.(1) Hover to touchdown: switch to inertial displays.	Return to CSM using PGNS.	(1) If failure occurs near end of powered descent (hover), mission completion is possible.
LR velocity data (display only)	No signals to display	Loss of crewman monitor capability	Prior to hover: instruct LGC to display velocity data on DSKY. Hover to touchdown: switch to inertial displays.		
LR range data (digital only)	Range data cannot be read into LGC	LGC cannot update its knowledge of range	Prior to hover: none.(1) Hover to touchdown: switch to inertial displays.		
LR range data (display only)	No signals to display	Loss of crewman monitor capability	Prior to hover: instruct LGC to display range data on DSKY. Hover to touchdown: switch to inertial displays.		
LR antenna position	Antenna cannot be put into position #2 (hover)	No LR data from hover to touchdown	Normal		Normally, switching LR antenna to position #2 is done at hover point. Usefulness of LR data from hover to touchdown is questionable. It appears that further study is required to determine if the LR requires a two position configuration.
LR data good signal (velocity or range)	Indicates failure when none exists	LGC will not receive LR data	Prior to hover: none.(1) Hover to touchdown: switch to inertial displays.		(1) If failure occurs near end of powered descent (hover), mission completion is possible.
Rendezvous Radar (RR) range, range rate, angle or angle rate. (digital only)	RR, range, range rate, angle or angle rate data cannot be read into LGC	LGC cannot update its knowledge of range, range rate, angle (2) or angle rate	Prior to initiation of powered descent: none. After start of powered descent: use CSM RR and VHF voice link. (1)		(1) Information derived from the CSM RR (range, range rate, angles and angle rates) must be conveyed to LEM via VHF voice link and inputed to LGC via DSKY. (2) The radar may be locked on and perfectly operational except for the indicated failure. This precludes the possibility of using the LGC for target acquisition or memory of target position when LEM maneuvers to such an extent that the target is outside the limits of angle coverage.
RR range, range rate, angle or angle rate data (display only)	No display signal	Loss of crewman monitor capability	Instruct LGC to display range, range rate, angle or angle rate on DSKY.		
CSM transponder	Transponder inoperative	Loss of LEM RR output except during transfer orbit insertion and final docking(1)	Prior to initiation of powered descent: none. After start of powered descent: use CSM RR and VHF voice link. (2)		(1) RR can skin track only at short range. (2) The CSM transponder should be operable from a command receiver since an incapacitated CSM crewman could not turn on the transponder. Therefore LEM would have to perform ascent without the use of any RR.

Primary Guidance and Navigation Subsystem (Cont.)

3.3 DESCRIPTION

The primary function of the guidance and navigation subsystem is to provide thrust and steering commands (guidance) to the LEM to accomplish lunar landing, ascent, rendezvous, and docking with the CSM. In order to accomplish these functions the G&N must obtain and act upon data to determine both the present LEM state vector and the CSM state vector (three position components and three velocity components) and to predict their future values based on a lunar model (navigation). The G&N Subsystem comprises the LEM Guidance Computer, the Inertial Measurement Unit, the Coupling Display Unit, the Power Servo Assembly, the Landing Radar, the Rendezvous Radar/Transponder, the Alignment Optical Telescope and controls and displays. Navigation data is also obtained by links with the CSM and Earth.

The G&N Subsystem is an aided free inertial system using celestial and radar fixes. The IMU is aligned to an inertial reference by star sights with the AOT. During descent to the lunar surface, inertial, altitude and velocity data from the LR are statistically weighted and mixed to provide improved navigation information. During the coasting descent, lunar stay and rendezvous phases of the mission, the RR coherently tracks its transponder on the CSM and provides range, range rate and inertial angular rate measurements to the LGC and astronaut displays.

3.3.1 LEM Guidance Computer

The LGC is the central data processor for the G&N Subsystem. It is a parallel, fixed point, whole word general purpose computer with a stored program. It has a rope core fixed memory and a ferrite core erasable memory. Programs may be coded in machine instructions or in interpreter instructions.

Computations are controlled by an Executive-Waitlist function. There is considerable interrupt and self checking capability.

Inputs to the LGC from the IMU, the LR and the RR are automatic. The astronaut may enter data via the Display and Keyboard; for example, data from the sightings with the AOT. The LGC performs four major functions:

1) Calculates steering signals and generates engine discretes necessary to meet trajectory requirements for the spacecraft.

2) Aligns the stable member of the IMU to a coordinate system defined by celestial measurements.

3) Conducts limited malfunction isolation within the G&N Subsystem, and

4) Displays pertinent navigation and mission information to the astronaut.

Using information from navigation fixes, the LGC determines the amount of deviation from a reference trajectory and calculates the necessary attitude and thrust corrective commands. Velocity corrections are measured by the IMU and controlled by the LGC. During coasting phases of the mission, velocity corrections are not made continuously but are initiated at predetermined check points.

3.3.2 Inertial Measurement Unit

The Inertial Measurement Unit (IMU) is the primary inertial sensing device of the LEM. Three integrating gyroscopes and three pendulous accelerometers are mounted on the innermost gimbal of a three-degree-of-freedom gimbal system. The innermost gimbal is held non-rotating with respect to inertial space by the three gimbal servos, which derive their input error signals from the three gyroscopes. The orthogonal input axes of the three accelerometers represent the Stable Member X, Y and Z axes.

Velocity changes along these axes are sensed by the accelerometers, which produce signals representing incremental changes in velocity and apply them to the LEM Guidance Computer for the calculation of total velocity. Signals proportional to the attitude of the LEM are generated by the inner, middle, and outer gimbal angle resolvers. These signals are applied to the LEM Guidance Computer and to the astronaut displays. The LEM Guidance Computer calculates gimbal angles that would result from bringing the LEM to a desired attitude, and compares these with the actual gimbal angles supplied by the gimbal angle resolvers. Any difference between the actual and calculated gimbal angles results in the generation of steering error signals, which are applied to the Stabilization and Control Subsystem to correct the attitude of the LEM.

3.3.3 Coupling Data Unit

The Coupling Data Units (CDU's) convert and transfer angular information between major elements of the G&N Subsystem and the Stabilization and Control Subsystem (SCS). The G&N Subsystem uses five CDU's one for the shaft axis, one for the trunnion axis of the Rendezvous Radar antenna and one for each of the three gimbals of the IMU.

The three CDU's working with the IMU provide interfaces between the IMU and the LEM Guidance Computer, and between the LEM Guidance Computer and the SCS. Each IMU gimbal angle resolver provides its CDU with analog gimbal angle signals representing the attitude of the LEM. The CDU converts these signals into digital form and applies them to the LEM Guidance Computer. Using these signals, the computer will calculate any necessary steering error signals and route them to the proper CDU in digital form. The CDU then converts these to 800 cps analog signals and applies them to the SCS. IMU coarse and fine align commands, which are generated by the LEM Guidance Computer, are coupled to the IMU by the CDU's.

The digital CDU principle is based upon the use of a-c analog computing techniques to derive error voltages from analog resolver angle signals. Digital circuitry continuously alters the analog computational scaling factor to null the error voltages. When this null occurs, the state of the digital memory drive circuit exactly corresponds to the resolver position.

3.3.4 Alignment Optical Telescope

The Alignment Optical Telescope (AOT) is a non-articulating, unit power, 60-degree field-of-view telescope used to determine the position and orientation of the spacecraft for aligning the Inertial Measurement Unit. The AOT provides the astronauts with data obtained by measuring angles between lines of sight to celestial bodies to establish the inertial reference. The telescope optics mechanism has three distinct viewing positions and a stowage position. Data obtained from the use of the AOT is manually entered into the LEM Guidance Computer.

3.3.5 Power and Servo Assembly

The Power and Servo Assembly (PSA) provides a central mounting place for the majority of the G&N Subsystem amplifiers, modular electronic components, and power supplies. The PSA consists of the following subassemblies: gimbal servo align and power amplifiers; gyro and accelerometer amplifiers and electronics; CDU electronics; power diodes and signal conditioners; and power supplies. A cold plate, through which water-glycol coolant from the Environmental Control Subsystem flows, is mounted under the PSA subassemblies to dissipate heat.

3.3.6 Landing Radar

The LR is composed of a low power transmitter receiver and data processor and an antenna assembly. The landing radar measures three orthogonal components of spacecraft velocity and range (along the altitude beam) by means of microwave contact with the lunar surface. The LR provides digital doppler and range data to the LGC for processing during the terminal portion of powered descent and the landing maneuver. This digital information is used to update the inertial data with respect to altitude and velocity. The LR also provides the same data for display purposes.

A strobe signal from the LGC is used to interrogate the LR for velocity and range data. The LGC also supplies the LR with a signal to change the antenna position from the descent position to the hover position. Antenna changeover occurs at an altitude of 200 feet.

Range and velocity data good signals are generated within the LR and are sent to the LGC. These signals are quantitative rather than qualitative and merely indicate that S/N is above a certain predetermined threshold.

The Landing Radar outputs are used for several important functions, including:

- Automatic update of inertial altitude at approximately 15, 000 feet.
- Automatic update of inertial spacecraft velocities from approximately 11, 000 feet.
- Visual display of velocity components to aid the astronauts to achieve hover.
- Visual display of velocity components and altitude to aid the astronauts in translation, letdown and touchdown.

3.3.7 Rendezvous Radar

The Rendezvous Radar is an amplitude comparison monopulse radar whose primary function is to provide data to the LGC to enable rendezvous with the CSM from a lunar surface launch or an aborted descent trajectory. The RR will track a transponder on the CSM while LEM is on the lunar surface, during the LEM coasting phases for monitoring or mid-course correction and during the Rendezvous maneuver. The RR can also skin or surface track for limited ranges. The RR supplies the LEM Guidance Computer with range, range rate and antenna angle information. The RR also provides both the aforementioned data and the angle rates for cockpit displays.

RR data and status to the LGC and commands from the LGC to the RR are transferred via a computer interface. The interface unit is composed of gating circuitry and a high speed counter. The LGC sends gate reset and gate strobe pulses to this unit to set the gating circuitry to steer selected RR data into the high speed counter. The LGC then sends counter readout pulses which transfer the RR data from the counter to the LGC input registers. The data is transferred to the LGC on two lines; the digital "ones" are sent on one line designated as "radar in 1" and the digital "zeros" are sent on the other line designated as "radar in 0". The descrete signals giving radar status are also sent to the LGC via the computer interface. The LGC receives antenna angle information and sends positioning commands to the RR through the interface unit.

Two CDU's, used with the Rendezvous Radar antenna, provide interfaces between the antenna and the LEM Guidance Computer. The LEM Guidance Computer calculates digital antenna position commands prior to acquisition of the Command Module transponder. These signals are converted into analog form

by the CDU's and applied to the antenna drive mechanism, causing the antenna to be aimed at the orbiting CSM. Following Transponder acquisition, CSM tracking information is digitized by the CDU's and supplied to the LEM Guidance Computer.

SECTION 4

STABILIZATION AND CONTROL SUBSYSTEM

4.1 SUMMARY

The SCS contingency analysis shows that feasible mission completion and abort modes are available for all failure cases except loss of the ATCA AC power supply.

Significant failures in the SCS include loss of individual or multiple jet commands, dead zone failures, rate feedback failures, and jet logic failures. Contingency modes that are available to compensate for such failures are the following:

- In the case of any horizontal jet command failing off (jets 3, 4 , 7, 8, 11, 12, 15 or 16), it is recommended that the jet pairs be shut off for either the mission completion or abort modes. Switching the logic avoids cross coupling, performance degradation and fuel penalty in all phases except rendezvous and docking. During these phases, a fuel penalty results due to compensation for the resulting unbalanced translational force. Fail off of a vertical jet command (TX jets 1, 2, 5, 6, 9, 10, 13 or 14) requires turn-off of the failed jet during all phases except when X-axis translation is required as in rendezvous. During X-axis translations, the compensation logic causes instability and therefore should not be used. Design changes in the logic are presently being considered. During ascent engine burn, switching out the failed jet initiates compensation which can account for large pitch moment unbalances. However, loss of a TX jet precludes unbalanced couple operation during ascent engine burn. The 213 fps normally derived from the RCS is unavailable during powered ascent. As a result, the CSM may be required to perform part of the final rendezvous maneuver and the docking.

- Single jet command fails on:

  When any jet command fails to turn off, it is necessary to manually turn off a jet pair. This shuts off fuel to the jet pair and switches the jet logic to obtain the most efficient use of the remaining jets. The penalty associated with loss of a jet pair (one vertical and one horizontal jet) is as described above.

- Two opposing jet commands fail off:

  In this case it is recommended that the system be left as-is for horizontal jet failures. Logic switching would result in loss of two vertical jets and would not improve translation capability

along the Y and Z axes. For vertical jet failures it is recommended that the appropriate jet pairs be shut off.

- Two opposing jet commands fail on:

  This constitutes a double failure and is not considered in the analysis. A single Pulse Ratio Modulator "on" failure, however, will cause opposing jets to operate in an ON-OFF mode. The modulator and jet gating become nothing more than an ON-OFF threshold. This mode of operation will result in higher fuel consumption since minimum impulse limit cycles will not be achieved. Vehicle response to attitude changes will not be greatly affected, (indeed, a failure of this type will probably be impossible to detect.) Dynamic studies are presently being performed to determine how much fuel penalty will result. If it is small, the best action will be to operate the system as-is.

- Loss of closed loop attitude control about any axis:

  Failures of this type may be due to dead band failures, RGA failures, or demodulator failures. The recommended contingency action consists of using the open loop "Pulse" or "Direct" modes to manually null the steering error indicators located within the FDAI. First phase man-machine simulations have shown that such a flight mode is feasible with little or no degradation in control performance. Crew task loading is heavy, however, a point that must be evaluated in terms of overall task loading. There is one failure which may result in mission failure and loss of the crew. This is failure of the ATCA AC power supply. This failure not only causes loss of closed loop attitude control, but loss of the displays needed for manual control as well. Design changes have been recommended.

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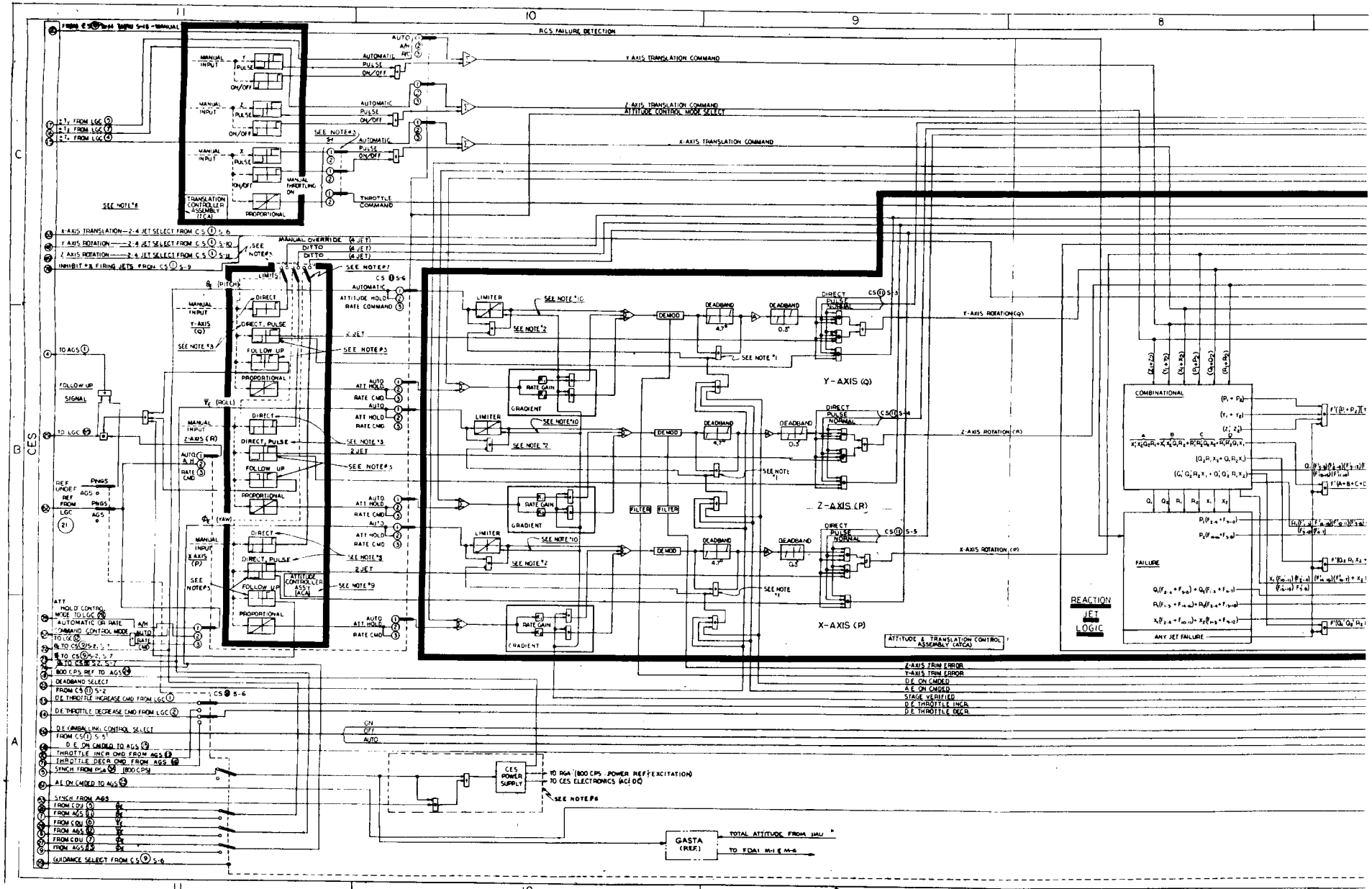

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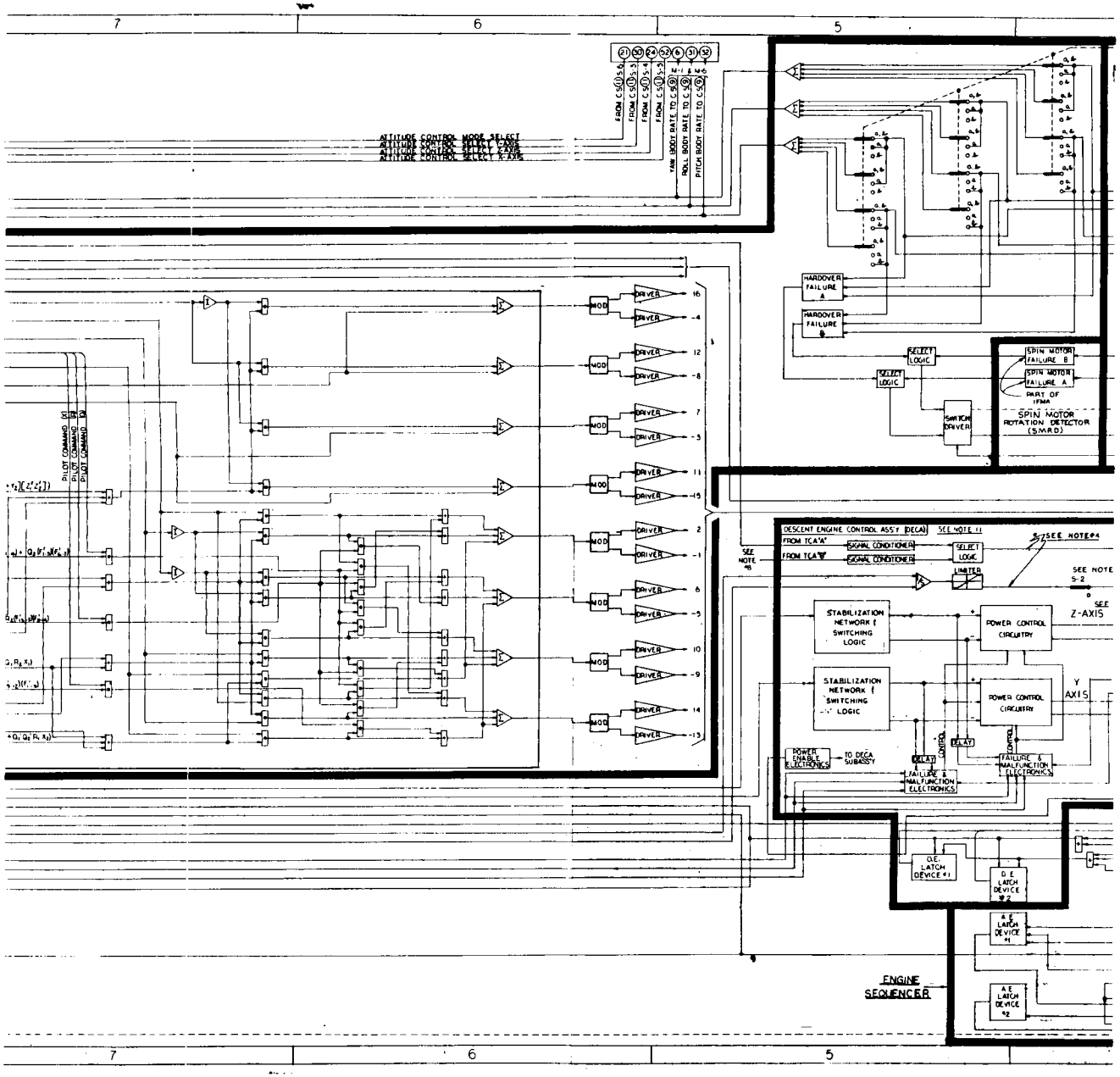
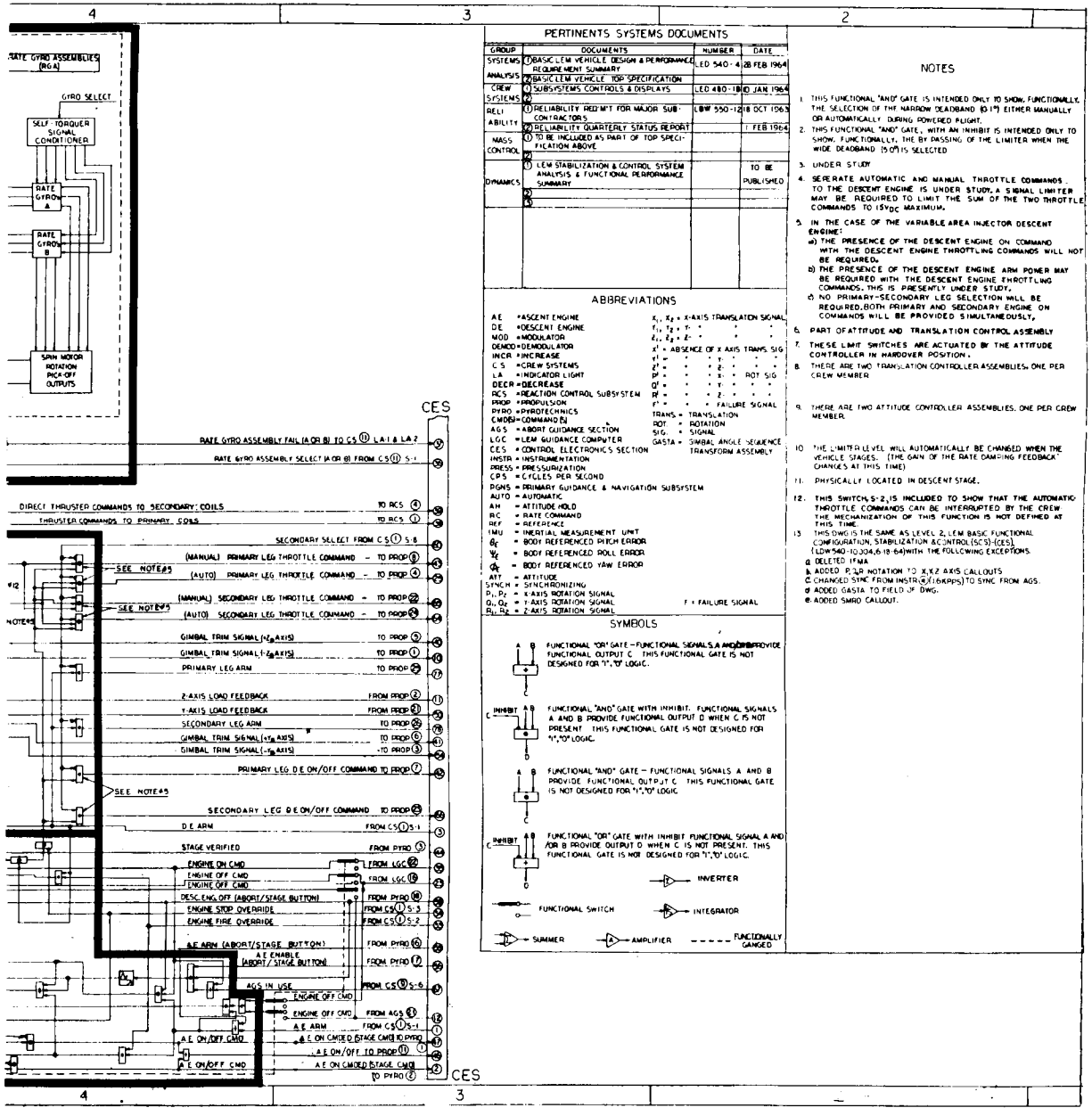


Figure 4-1 Functional Config


4
3
2
RATE GYRO ASSEMBLIES (RGA)
GYRO SELECT
SELF-TORQUER SIGNAL CONDITIONER
RATE GYRO A
RATE GYRO B
SPIN MOTOR ROTATION PICK-OFF OUTPUTS
CES
RATE GYRO ASSEMBLY FAIL (A OR B) TO CS (11) L A1 & LA2
RATE GYRO ASSEMBLY SELECT (A OR B) FROM CS (11) S-1
DIRECT THRUSTER COMMANDS TO SECONDARY COILS TO RCS
THRUSTER COMMANDS TO PRIMARY COILS TO RCS
SECONDARY SELECT FROM CS (1) S-8
(MANUAL) PRIMARY LEG THROTTLE COMMAND – TO PROP
(AUTO) PRIMARY LEG THROTTLE COMMAND – TO PROP
(MANUAL) SECONDARY LEG THROTTLE COMMAND – TO PROP
(AUTO) SECONDARY LEG THROTTLE COMMAND – TO PROP
GIMBAL TRIM SIGNAL (+Z_B AXIS) TO PROP
GIMBAL TRIM SIGNAL (-Z_B AXIS) TO PROP
PRIMARY LEG ARM TO PROP
Z-AXIS LOAD FEEDBACK FROM PROP
Y-AXIS LOAD FEEDBACK FROM PROP
SECONDARY LEG ARM TO PROP
GIMBAL TRIM SIGNAL (+Y_B AXIS) TO PROP
GIMBAL TRIM SIGNAL (-Y_B AXIS) TO PROP
PRIMARY LEG D.E. ON/OFF COMMAND TO PROP
SECONDARY LEG D.E. ON/OFF COMMAND TO PROP
D.E. ARM FROM CS (1) S-1
STAGE VERIFIED FROM PYRO
ENGINE ON CMD FROM LGC
ENGINE OFF CMD FROM LGC
DESC. ENG. OFF (ABORT/STAGE BUTTON) FROM PYRO
ENGINE STOP OVERRIDE FROM CS (1) S-3
ENGINE FIRE OVERRIDE FROM CS (1) S-2
A.E. ARM (ABORT/STAGE BUTTON) FROM PYRO
A.E. ENABLE (ABORT/STAGE BUTTON) FROM PYRO
AGS IN USE FROM CS (9) S-6
ENGINE OFF CMD FROM AGS
A.E. ARM FROM CS (1) S-1
A.E. ON CMD'D (STAGE CMD) TO PYRO
A.E. ON/OFF TO PROP
A.E. ON CMD'D (STAGE CMD) TO PYRO
A.E. ON/OFF CMD
A.E. ON/OFF CMD
SEE NOTE 5
SEE NOTE 5
SEE NOTE 5
NOTE 5
#12

PERTINENTS SYSTEMS DOCUMENTS

GROUP		DOCUMENTS	NUMBER	DATE
SYSTEMS ANALYSIS	(1)	BASIC LEM VEHICLE DESIGN & PERFORMANCE REQUIREMENT SUMMARY	LED 540-4	28 FEB 1964
	(2)	BASIC LEM VEHICLE TOP SPECIFICATION		
CREW SYSTEMS	(1)	SUBSYSTEMS CONTROLS & DISPLAYS	LED 480-18	10 JAN 1964
RELIABILITY	(1)	RELIABILITY REQMT FOR MAJOR SUB-CONTRACTORS	LSP 550-12	18 OCT 1963
	(2)	RELIABILITY QUARTERLY STATUS REPORT		1 FEB 1964
MASS CONTROL	(1)	TO BE INCLUDED AS PART OF TOP SPECIFICATION ABOVE		
	(2)			
DYNAMICS	(1)	LEM STABILIZATION & CONTROL SYSTEM ANALYSIS & FUNCTIONAL PERFORMANCE SUMMARY		TO BE PUBLISHED
	(2)			
	(3)			

ABBREVIATIONS

AE	*ASCENT ENGINE	X_1, X_2	= X-AXIS TRANSLATION SIGNAL
DE	*DESCENT ENGINE	Y_1, Y_2	= Y- " " "
MOD	*MODULATOR	Z_1, Z_2	= Z- " " "
DEMOD	*DEMODULATOR	X^1	= ABSENCE OF X AXIS TRANS. SIG
INCR	*INCREASE	Y^1	= " " Y- " " "
CS	*CREW SYSTEMS	Z^1	= " " Z- " " "
LA	*INDICATOR LIGHT	P^1	= " " S_1- " ROT SIG
DECR	*DECREASE	Q^1	= " " Y- " " "
RCS	*REACTION CONTROL SUBSYSTEM	R^1	= " " Z- " " "
PROP	*PROPULSION	F^1	= " " FAILURE SIGNAL
PYRO	*PYROTECHNICS	TRANS	= TRANSLATION
CMD(S)	=COMMAND(S)	ROT	= ROTATION
AGS	= ABORT GUIDANCE SECTION	SIG	= SIGNAL
LGC	= LEM GUIDANCE COMPUTER	GASTA	= SIMBAL ANGLE SEQUENCE TRANSFORM ASSEMBLY
CES	= CONTROL ELECTRONICS SECTION		
INSTR	= INSTRUMENTATION		
PRESS	= PRESSURIZATION		
CPS	= CYCLES PER SECOND		
PGNS	= PRIMARY GUIDANCE & NAVIGATION SUBSYSTEM		
AUTO	= AUTOMATIC		
AH	= ATTITUDE HOLD		
RC	= RATE COMMAND		
REF	= REFERENCE		
IMU	= INERTIAL MEASUREMENT UNIT		
θ_E	= BODY REFERENCED PITCH ERROR		
ψ_E	= BODY REFERENCED ROLL ERROR		
ϕ_E	= BODY REFERENCED YAW ERROR		
ATT	= ATTITUDE		
SYNCH	= SYNCHRONIZING		
P_1, P_2	= X-AXIS ROTATION SIGNAL		
Q_1, Q_2	= Y-AXIS ROTATION SIGNAL	F	= FAILURE SIGNAL
R_1, R_2	= Z-AXIS ROTATION SIGNAL		

SYMBOLS

FUNCTIONAL "OR" GATE – FUNCTIONAL SIGNALS A AND B PROVIDE FUNCTIONAL OUTPUT C. THIS FUNCTIONAL GATE IS NOT DESIGNED FOR "1","0" LOGIC.

FUNCTIONAL "AND" GATE WITH INHIBIT. FUNCTIONAL SIGNALS A AND B PROVIDE FUNCTIONAL OUTPUT D WHEN C IS NOT PRESENT. THIS FUNCTIONAL GATE IS NOT DESIGNED FOR "1","0" LOGIC.

FUNCTIONAL "AND" GATE – FUNCTIONAL SIGNALS A AND B PROVIDE FUNCTIONAL OUTPUT C. THIS FUNCTIONAL GATE IS NOT DESIGNED FOR "1","0" LOGIC

FUNCTIONAL "OR" GATE WITH INHIBIT FUNCTIONAL SIGNAL A AND/OR B PROVIDE OUTPUT D WHEN C IS NOT PRESENT. THIS FUNCTIONAL GATE IS NOT DESIGNED FOR "1","0" LOGIC.

INVERTER

FUNCTIONAL SWITCH

INTEGRATOR

SUMMER

AMPLIFIER

FUNCTIONALLY GANGED

NOTES

1. THIS FUNCTIONAL "AND" GATE IS INTENDED ONLY TO SHOW, FUNCTIONALLY, THE SELECTION OF THE NARROW DEADBAND (0.1°) EITHER MANUALLY OR AUTOMATICALLY DURING POWERED FLIGHT.
2. THIS FUNCTIONAL "AND" GATE, WITH AN INHIBIT IS INTENDED ONLY TO SHOW, FUNCTIONALLY, THE BY PASSING OF THE LIMITER WHEN THE WIDE DEADBAND (5.0°) IS SELECTED
3. UNDER STUDY
4. SEPERATE AUTOMATIC AND MANUAL THROTTLE COMMANDS TO THE DESCENT ENGINE IS UNDER STUDY. A SIGNAL LIMITER MAY BE REQUIRED TO LIMIT THE SUM OF THE TWO THROTTLE COMMANDS TO 15 V_{DC} MAXIMUM.
5. IN THE CASE OF THE VARIABLE AREA INJECTOR DESCENT ENGINE:
   a) THE PRESENCE OF THE DESCENT ENGINE ON COMMAND WITH THE DESCENT ENGINE THROTTLING COMMANDS WILL NOT BE REQUIRED.
   b) THE PRESENCE OF THE DESCENT ENGINE ARM POWER MAY BE REQUIRED WITH THE DESCENT ENGINE THROTTLING COMMANDS. THIS IS PRESENTLY UNDER STUDY.
   c) NO PRIMARY-SECONDARY LEG SELECTION WILL BE REQUIRED, BOTH PRIMARY AND SECONDARY ENGINE ON COMMANDS WILL BE PROVIDED SIMULTANEOUSLY.
6. PART OF ATTITUDE AND TRANSLATION CONTROL ASSEMBLY
7. THESE LIMIT SWITCHES ARE ACTUATED BY THE ATTITUDE CONTROLLER IN HANDOVER POSITION.
8. THERE ARE TWO TRANSLATION CONTROLLER ASSEMBLIES, ONE PER CREW MEMBER
9. THERE ARE TWO ATTITUDE CONTROLLER ASSEMBLIES, ONE PER CREW MEMBER
10. THE LIMITER LEVEL WILL AUTOMATICALLY BE CHANGED WHEN THE VEHICLE STAGES. (THE GAIN OF THE RATE DAMPING FEEDBACK CHANGES AT THIS TIME)
11. PHYSICALLY LOCATED IN DESCENT STAGE.
12. THIS SWITCH, S-2, IS INCLUDED TO SHOW THAT THE AUTOMATIC THROTTLE COMMANDS CAN BE INTERRUPTED BY THE CREW. THE MECHANIZATION OF THIS FUNCTION IS NOT DEFINED AT THIS TIME.
13. THIS DWG IS THE SAME AS LEVEL 2, LEM BASIC FUNCTIONAL CONFIGURATION, STABILIZATION & CONTROL (SCS)-(CES), (LDW 540-10 304, 6-18-64) WITH THE FOLLOWING EXCEPTIONS.
   a. DELETED 1FMA
   b. ADDED P_2,R_2 NOTATION TO X,Y,Z AXIS CALLOUTS
   c. CHANGED SYNC FROM INSTR (6)(16KPPS) TO SYNC FROM AGS.
   d. ADDED GASTA TO FIELD OF DWG.
   e. ADDED SMRD CALLOUT.

;uration, Stabilization and Control Subsystem, Control Electronics Section

SUBSYSTEM: Stabilization and Control | 4.2 WORKSHEETS | *PHASE:* All LEM Phases

EQUIPMENT/ FUNCTION	*FAILURE MODE*	*UNCOMPENSATED FAILURE EFFECT*	*MISSION COMPLETION MODE*	*MISSION ABORT MODE*	*REMARKS*
ATCA command to single vertical thruster (QRX)	Off		Leave system as-is for X translation and rendezvous maneuvers. Switch jet out during all other phases.	Same as mission completion mode.	In phases requiring X-axis translation, instability results when compensation is used. A change in the logic to correct this situation is being recommended. During powered ascent, large Q unbalance with no compensation results in instability in pitch rate.
ATCA command to single horizontal thruster (PYZ)	Off		Switch out failed jet and continue.		Switching logic avoids all cross-coupling effects. No performance degradation or fuel penalty results except for the following: jet failed is one that is normally used for translation (such as in rendezvous and docking). In this case there will be a fuel penalty resulting from the unbalanced translational force.
ATCA command to any thruster	On		Close appropriate isolation valve to shut off thruster pair. Logic automatically selects proper alternate thrusters. To illustrate, if thruster 1 fails, thrusters 1 and 3 are shut off. Commanded - Z translation will also cause a positive pitch rotation. This is automatically compensated for by a negative pitch tongue.		Automatic failure sensing is being studied. If steady-state attitude error reaches 0.7 deg in any axis during powered ascent, crew must switch to balanced couples for attitude control.
ATCA commands to two opposing vertical thrusters (QRX)	Off		Close appropriate isolation valve to shut off both thruster pairs. Logic automatically selects proper alternate thrusters		A small fuel penalty results. The action is necessary to prevent instability and to overcome large moment unbalances.
ATCA commands to two opposing horizontal thrusters (PYZ)	Off		Leave system as-is.		See remarks on failure of ATCA command to single horizontal thruster.
ATCA commands to any two opposing thrusters	PRM fails to on-off operation only		Close appropriate isolation valve to shut off both thruster pairs. Logic automatically selects proper alternate thrusters.		Being studied. Failure results in on-off control rather than pulse-ratio modulated control. It may be preferable to leave the system as-is in certain phases.
ATCA pitch (Q) or roll (R) ±0.3 deg deadband	Fails to ±5 deg. (1) or One or both sides fail open		• Descent orbit insertion: None. • Powered Descent: Continue normally (2) with 5-deg deadband, if available, or switch to direct mode and fly error needle in failed axis. • Manual landing: Switch to direct mode in failed axis (3). Powered ascent: Same as powered descent. • Midcourse corrections: Use direct mode in failed axis and fly error needle. • Docking: Use pulse mode in failed axis.		(1) Assumes attempt to switch deadband manually does not work. (2) Powered flight with 5-deg deadband using PGNS appears feasible; is being studied. Feasibility of direct mode with error needles has been demonstrated in Phase A simulations. (3) Feasibility demonstrated in Phase A simulations.
ATCA yaw (P) ± 0.3 deg deadband	Same as above		Continue mission with 5-deg deadband or pulse mode. Must use pulse mode for alignment phases. Maintain attitude within 2-5 deg during powered flight 10-20 deg during coasting flight		
ATCA pitch (Q), roll (R), or yaw (P) 5-deg deadband	Fails to ± 0.3 deg, or one or both sides fail open (1)		Switch to pulse mode in failed axis during coasting. Maintain attitude within 10-20 deg to conserve RC fuel.		Crew task loading to be studied (1) The 5-deg deadband is bypassed in powered flight.

SUBSYSTEM: Stabilization and Control

PHASE: All LEM Phases

EQUIPMENT/ FUNCTION	*FAILURE MODE*	*UNCOMPENSATED FAILURE EFFECT*	*MISSION COMPLETION MODE*	*MISSION ABORT MODE*	*REMARKS*
Rate signal from any gyro in one RGA package	Hard over		Continue mission in all modes. Automatic failure-detection circuit switches out failed package and failure light goes on.	Same as mission completion mode.	For failure in both packages, pilot can select either RGA package and fly failed axis in direct mode.
	High zero offset or standoff signal, or no signal		No auto failure detection. Pilot must sense attitude error from error needles and switch out failed package (hit and miss).	Same as mission completion mode.	
	Loss of spin motor		Continue mission in all modes. Spin motor rotation detector (SMRD) switches automatically to other package.	Same as mission completion mode.	SMRD will also switch automatically if gyro motor speed decreases below a certain level.
Pitch (θ_e), roll (ψ_e) or yaw (ϕ_e) steering error from GCA or PGNS	Open or bias; error needle operative		Automatic phases: Switch to direct 1 mode in failed axis and fly error needle (1). Coasting phases and normal landing: Use rate command mode or switch to AGS to obtain attitude hold mode.	Descent orbit insertion: Shutoff DE, switch to AGS attitude readout, and rendezvous and dock manually using attitude hold mode. Powered descent or ascent: Switch to AGS and press abort button. All phases: If failure also affects AGS steering, switch to direct mode in failed axis and fly error needle.	(1) Simulation studies are being performed to determine if Phase II can be flown completely manually with the attitude hold mode.
Attitude Controller Assembly (ACA) pulse/direct switches for each axis	Stuck open	Not able to command an attitude change	Pilot must use other controller or proportional output.	Same as mission completion mode.	
	Stuck closed	Will command a pulsed or two-thruster direct attitude change.	Pilot must remove ACA connector and use other controller.		
ACA manual override switch (four-thruster direct switch)	Stuck open	RCS thrusters will not fire when command is given.	Pilot sense no change and uses other controller or different mode.		This switch not normally used.
	Stuck closed	RCS thrusters will fire continuously	Pilot must remove ACA connector and use other controller.		Same as above.
ACA pitch, roll, or yaw rate command	Off		Use alternate controller		
	Hard over, bias, or noisy		Disconnect defective controller and use alternate controller.		
ACA follow up switch	Open	Does not hold attitude	Use alternate controller.		
Detent switch	Closed	Cannot command attitude rate	Disconnect defective controller and use alternate controller.		
Translation Controller Assembly (TCA) A_x, A_y, or A_z	Pulse mode or on/off mode failed off		Use alternate controller.		
	Pulse mode or on/off mode failed on		Disconnect defective controller and use alternate TCA.		
TCA throttle command	Open or shorted		Switch to alternate TCA.		

SUBSYSTEM: Stabilization and Control

PHASE: All LEM Phases

EQUIPMENT/ FUNCTION	*FAILURE MODE*	*UNCOMPENSATED FAILURE EFFECT*	*MISSION COMPLETION MODE*	*MISSION ABORT MODE*	*REMARKS*
Total attitude signals from AGS	Large error		Use PGNS.	Use PGNS.	This output can be checked against PGNS during mission. Verify that error is in AGS by visual observation of lunar surface, sun, or earth.
	Small error		Use PGNS.	Use PGNS.	Verify that error is in AGS by commanding pure P, Q, and R rotations and observing 8-ball motion. This method may be used in coasting flight only. Reference platform that failed will probably give erratic motion on 8-ball using this test.
800-cycle synch from PGNS	Off, or out of synch		Prior to manual landing: None During manual landing: Switch to AGS for attitude reference.	Switch to AGS	
800-cycle synch to GASTA	Off, or out of synch		Use AGS for total attitude for normal landing.	Use PGNS (2).	One FDAI is driven by PGNS via GASTA; the other by AGS directly.
Total attitude signals from GASTA	Any		Same as above.	Same as above.	Verify that failure is in GASTA by checking one 8-ball against the other.
±15 vdc power output from ATCA power supply	Off		If landing is imminent, can continue using PGNS to drive 8-ball and direct mode in all three axes for attitude control. Ascent must be performed manually using error needles and direct mode. Sequencing must also be done manually.	Abort with PGNS, using error needles and direct mode in all three axes. Sequencing must also be done manually.	Failure results in no forward path in attitude control loops and loss of SCS logic. Phase A simulations show these modes feasible; need final verification in Phase B simulation. ATCA power supply designed for extremely high reliability.
±4 vdc power output from ATCA power supply	Off		Switch to unbalanced couples for all powered phases (no logic required).	Same as mission completion mode.	Failure results in loss of DECA logic, RC thruster logic, SMRD logic, and sequencer logic.
Three- or single-phase AC power from ATCA power supply	Off		None.	None.	AC power needed for RGA, displays, TC, AC, and ATCA. Cannot fly automatic or direct mode without rate information. Design change to be recommended (possibly redundant power supply). Must verify in Phase B simulations that direct mode cannot be flown without rate information.

SUBSYSTEM: Stabilization and Control

PHASE: All Descent Phases

EQUIPMENT/ FUNCTION	*FAILURE MODE*	*UNCOMPENSATED FAILURE EFFECT*	*MISSION COMPLETION MODE*	*MISSION ABORT MODE*	*REMARKS*
Pitch or roll gimbal trim signal from DECA or ATCA to actuator	Off or on.		• Descent orbit insertion: None (1). • Powered descent: Continue as is until either steady-state attitude error exceeds approximately 0.6 deg, or RC fuel useage exceeds budget for descent phase.	• Prior to powered descent: Abort using RC thrusters. • Powered descent: Abort with descent engine until steady-state attitude error exceeds approximately 0.6 deg, or RC fuel useage reaches time-related fuel budget; then stage and use ascent engine.	(1) Initial c-g offset at powered descent could be catastrophic.
DE ON signal from GCA or DECA	Fails to appear		• Descent orbit insertion or powered descent: Turn on manually.	• Lunar orbit: Switch to attitude hold mode and redock. • Powered descent: Press abort button. PGNS commands abort with RC thrusters.	Effects of delay being studied.
DE OFF signal from GCA or DECA	Fails to appear		• Descent orbit insertion or lunar touchdown: Turn off manually (may be manual normally for lunar touchdown).	• Descent orbit insertion: Turn off manually and press abort button. PGNS commands abort with RC thrusters. • Lunar touchdown: Not applicable.	Effects of delay being studied. Fail-safe circuitry keeps descent engine on when both on and off commands are preset. The completion and abort modes also apply if DE on signal fails to disappear upon turn-off command.
Auto-throttle command from GCA or DECA	Off, hard over, or erratic (1)		• Descent orbit injection or Phase I powered descent: None. • Phase II powered descent: Switch to attitude hold mode and fly manually using T/W indicator, velocity, FDAI indicators, and visual. (2). • Ascent phases: Not applicable.	• Descent orbit insertion: Manually shut-off DE and press abort button. • Phase I or II powered descent: Press abort-stage button. • Ascent phases: Not applicable.	(1) Can determine if failure occurs before propulsion interface by observing auto throttle indicator. (2) Feasibility of completely manual descent orbit insertion and powered descent being studied on simulator.
Manual throttle command from DECA	Any		• During all descent phases: None. • Ascent phases: Not applicable.	All descent phases: Abort-stage.	Assumes failure in DECA after or-gating of the two manual throttle commands. For failures before or-gate, see TC failures.
DE ARM signal from DECA	Off		None.	• Lunar orbit: Switch to attitude hold mode and redock. • Just prior to powered descent: Abort with RCS using PGNS. • During powered descent: Abort-stage.	

SUBSYSTEM: Stabilization and Control

PHASE: All Ascent Phases

EQUIPMENT/ FUNCTION	*FAILURE MODE*	*UNCOMPENSATED FAILURE EFFECT*	*MISSION COMPLETION MODE*	*MISSION ABORT MODE*	*REMARKS*
AE ON signal from GCA or Eng Seq	Fails to appear		Lunar launch: Turn on manually.	Not applicable.	
AE ARM signal (manual) from GCA or Eng Seq	Off		Lunar launch: Press abort-stage button to provide separate arm signal.	Not applicable.	
AE OFF signal from GCA or Eng Seq	Fails to appear		Turn off manually.	Not applicable.	Effects of delay should be studied.
GCA auto translation commands (1)	On		Ascent midcourse corrections and rendezvous: Manually switch to attitude hold mode (2).	Same as mission completion mode.	(1) Automatic translation used only for midcourse corrections and rendezvous. (2) Feasibility demonstrated by Phase A simulation.
	Off		Leave in auto mode and translate manually using ΔV indicator (2).		Same as above.

4.3 DESCRIPTION

The Stabilization and Control Subsystem (SCS) is divided into two major sections: Control Electronics Section (CES) and Abort Guidance Section (AGS) (see Figure 4-1). The CES consists of a control panel, an attitude and translation control assembly (ATCA), an engine sequencer, a descent engine control assembly (DECA), two gimbal drive actuator assemblies (GDA), two attitude controller assemblies (ACA), two translation controller assemblies (TCA), and two rate gyro assemblies (RGA). The AGS implementation is presently in a preliminary development stage. Therefore a detailed equipment description is not available at this time.

4.3.1 Control Electronics Section

The CES converts attitude error, rate commands, or translation command signals generated in the Guidance and Navigation (G & N) Subsystem, the AGS, or manually operated cockpit controls into pulse-ratio-modulated pulsed or full-on signals for firing the thrust chambers in the Reaction Control Subsystem (RCS). Rate and attitude error signals from the CES are routed to displays; the logic circuits within the CES route the signals to the appropriate thrusters. In addition, the CES routes on-off commands for the ascent and descent engines, and conditions automatic and manual throttle commands to the descent engine. Trim control of the gimballed descent engine is also provided to assure that the thrust vector operates through the LEM vehicle's center of gravity.

4.3.1.1 Attitude Control: Five basic modes of attitude control are provided by the CES and are selectable on the control panel. They are: automatic mode, attitude hold mode, rate command mode, pulse mode and direct mode. The pulse and direct modes are selectable on an individual axis basis.

4.3.1.1.1 Automatic Mode: The automatic mode provides fully automatic attitude control. When this mode is selected, attitude error signal from the G & N Subsystem or the AGS are applied to the CES. The attitude error signals are routed to the ATCA, where they are combined with angular rate signals from the RGA. These signals are then demodulated and applied through deadband circuits, logic circuits, pulse modulators, and jet driver amplifiers to fire the appropriate RCS thrust chambers. Pitch and roll signals are also routed to the DECA for automatic trim control of the descent engine after demodulation. The selectable deadband feature provides a wide and narrow deadband (MAX and MIN). The wide deadband is used during coasting phases of the mission to conserve fuel; the narrow deadband is used when accurate thrust vector control is required. The signals applied to the logic circuit are processed and directed to the appropriate pulse modulators and jet driver amplifiers. The pulse modulators and jet drivers generate the fire pulses that actuate the RCS thrust chambers.

4.3.1.1.2 Attitude Hold Mode: The attitude hold mode is the primary attitude control mode for the hover, landing, and docking phases of the mission. When this mode is selected, the pilot manually commands attitude change rates by displacing the attitude controller from the neutral position. When the controller is in neutral, the error signals from the G & N Subsystem or AGS causes the LEM to hold attitude.

4.3.1.1.3 Rate Command Mode: The rate command mode functions in the same manner as the attitude hold mode except that attitude hold is not provided when the attitude controller is in the neutral position.

4.3.1.1.4 Pulse Mode: The pulse mode is an open loop attitude control mode. In this mode small acceleration (pulsed) commands can be made about the selected axis using the attitude controller (the remaining axes operate in the mode selected on the mode select switch).

4.3.1.1.5 Direct Mode: The direct mode is also an open loop attitude control mode. In this mode two thruster acceleration commands can be made about the selected axis using the attitude controller (the remaining axes operate in the mode selected on the mode select switch).

4.3.1.1.6 Manual Override: Manual override can be activated in any mode by displacing the attitude controller to its full travel. This provides four thruster operation through the secondary solenoids of the RCS thrust chambers which are directly wired to switches in the attitude controller for maximum reliability.

4.3.1.2 Translation Control: Both automatic and manual translation control using the reaction jets are provided. Automatic translation control consists of on-off commands from the G & N Subsystem. These commands are connected only when the mode switch is in automatic and are used for small velocity changes and out-of-plane correction during the mission. Both pulse and on-off manual translation control are provided in all three axes for docking and for small corrections during hover and landing. Detent switches in the translation controller insert the proper voltage level into the logic for operation of the RCS jets.

4.3.2 Abort Guidance Section

Three major functions are provided by this section: abort capabilities from any point in power descent or ascent; attitude reference for vehicle stabilization; and capabilities for ascent from the lunar surface. The AGS generates LEM attitude error signals that, after passing through the CES, actuate appropriate RCS thrusters. Additional functions are provided in the AGS by the IFMA. These functions are: monitoring the spin motor rotation of each gyro in the RGA; stimulating the SCS for OBCE (part of Instrumentation Subsystem) tests; and providing unique signal conditioning for SCS's signals to be telemetered. The AGS has three modes: warmup, alignment, and inertial reference. A functional configuration drawing is not available at this time.

4.3.2.1 Warm-up Mode: The warm-up mode provides 28-volts d-c power to the AGS to prepare the AGS for operation 30 minutes after entering this mode.

4.3.2.2 Alignment Mode: The alignment mode has three sub-modes: IMU alignment, orbital alignment, and lunar alignment of the AGS. The IMU alignment is accomplished by aligning the AGS Euler angles to the IMU gimbal angles. Orbital alignment is used if the G & N Subsystem malfunctions and the LEM vehicle is in lunar orbit for such extended periods that the subsystems must be shutdown to conserve power. Orbital alignment is accomplished by manually achieving the proper LEM vehicle attitude with respect to the Command/Service Module's orbit, and manually pitching the LEM vehicle to a predetermined pitch Euler angle. Lunar alignment is used in preparation for ascent from the lunar surface if the G & N Subsystem has malfunctioned. Lunar alignment is accomplished by manually setting the yaw Euler angle to the same value obtained from the AGS control panel after lunar landing. The pitch and roll Euler angles are determined by local vertical sensors.

4.3.2.3 Inertial Reference Mode: The inertial reference mode has four sub-modes: followup, abort attitude hold, automatic, and manual. The followup sub-mode permits the AGS to follow LEM vehicle motion to prevent transients when control is switched from the G & N Subsystem to the AGS (LEM attitude error signal equal to zero). When the AGS is selected as the inertial reference, the LEM vehicle enters the abort attitude hold sub-mode and maintains the last attitude it held prior to the switch-over. The automatic sub-mode provides automatic preprogrammed abort trajectories from any point in the lunar descent or ascent, or a preprogrammed ascent trajectory from the lunar surface. In manual sub-mode, the capability is provided to allow the astronaut to accurately control the LEM's attitude and provide manual abort capabilities.

SECTION 5

ELECTRICAL POWER/POWER GENERATION SECTION

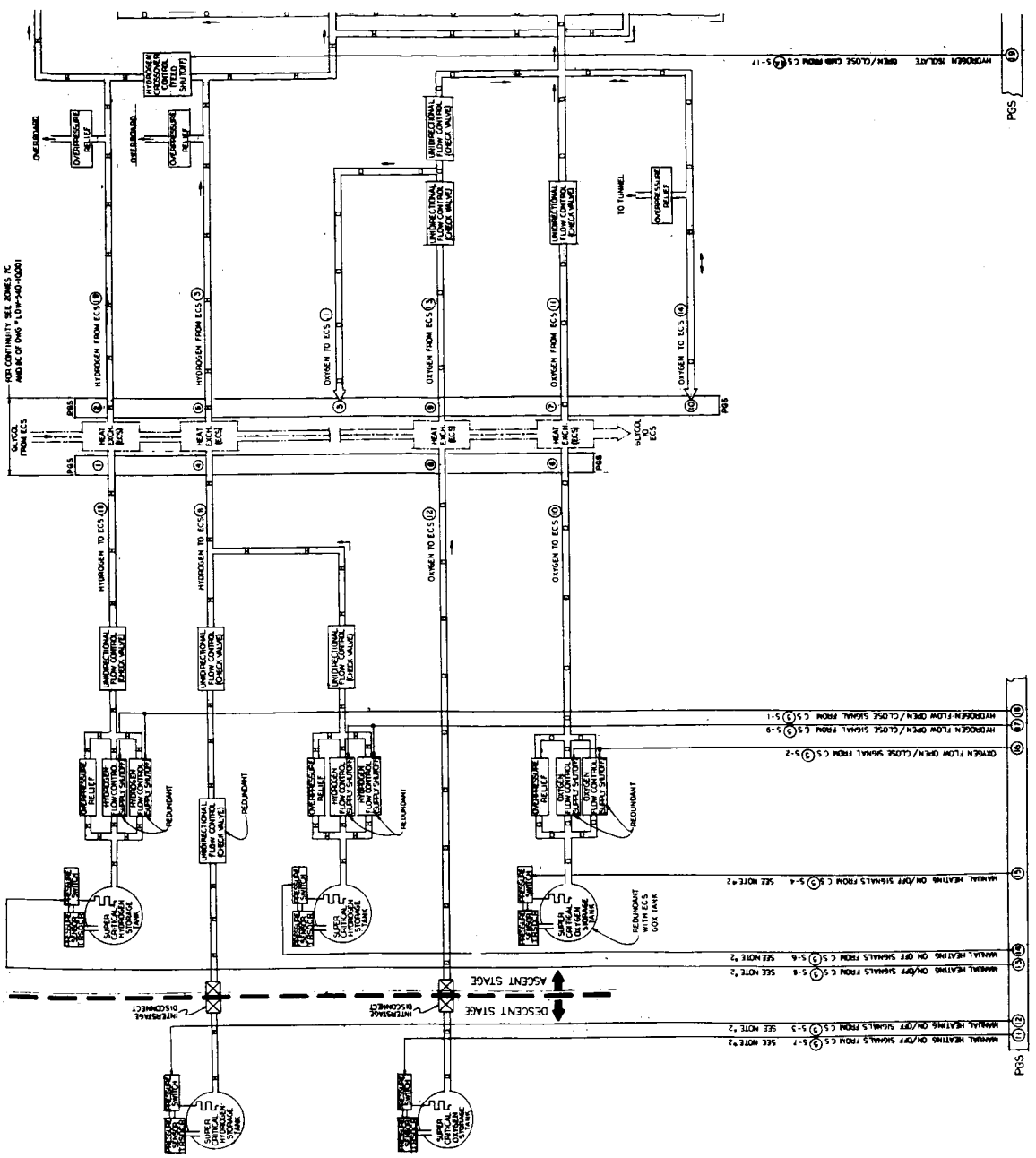
5.1 SUMMARY

Contingency analysis of the PGS indicates that:

- For any single failure, sufficient power and energy are available to permit abort under survival load conditions in any mission phase, although backup and orbital contingency capabilities may not be available.
- For failure of a single fuel cell assembly, descent cryogenic tank (after lunar separation) or ascent cryogenic tank, sufficient power and energy are available for performance of a shortened mission, although, as above, backup and orbital contingency capability may not be available.
- Failure of hydrogen or oxygen interstage disconnect to separate would result in loss of the crew. Single disconnects have been utilized because of the predicted high reliability of these devices.
- Operational uses of the EPS - ECS shut-off valve are not sufficiently defined at present to permit a definitive analysis of the effect of tank failure. This area should be restudied when the current PGS redesign effort is completed.

The hydrogen tank and the oxygen tank in the LEM descent stage contain the reactants to allow the FCA's to provide power to the LEM during Launch Countdown. Translunar, Lunar Descent and Lunar Stay time. In addition, oxygen required for the ECS is also contained in the oxygen storage tanks. Upon staging, the ascent oxygen tank and the two ascent stage hydrogen tanks provide the reactants for operating the ECS and each FCA. The GOX tank will provide redundancy for the ascent oxygen tank.

5.1.3.A


HYDROGEN FROM ECS
OXYGEN TO ECS
OXYGEN FROM ECS
HYDROGEN TO ECS
GLYCOL FROM ECS
GLYCOL TO ECS
HEAT EXCH ECS
OVERPRESSURE RELIEF
UNIDIRECTIONAL FLOW CONTROL (CHECK VALVE)
TO TUNNEL
HYDROGEN FLOW CONTROL (SHUT OFF)
OXYGEN FLOW CONTROL (SHUT OFF)
REDUNDANT
REDUNDANT WITH ECS GOX TANK
SUPER CRITICAL HYDROGEN STORAGE TANK
SUPER CRITICAL OXYGEN STORAGE TANK
PRESSURE SENSOR
PRESSURE SWITCH
ASCENT STAGE
DESCENT STAGE
INTERSTAGE DISCONNECT
PGS
MANUAL HEATING ON/OFF SIGNALS FROM C&S
SEE NOTE 2
HYDROGEN FLOW OPEN/CLOSE SIGNAL FROM C&S
OXYGEN FLOW OPEN/CLOSE SIGNAL FROM C&S

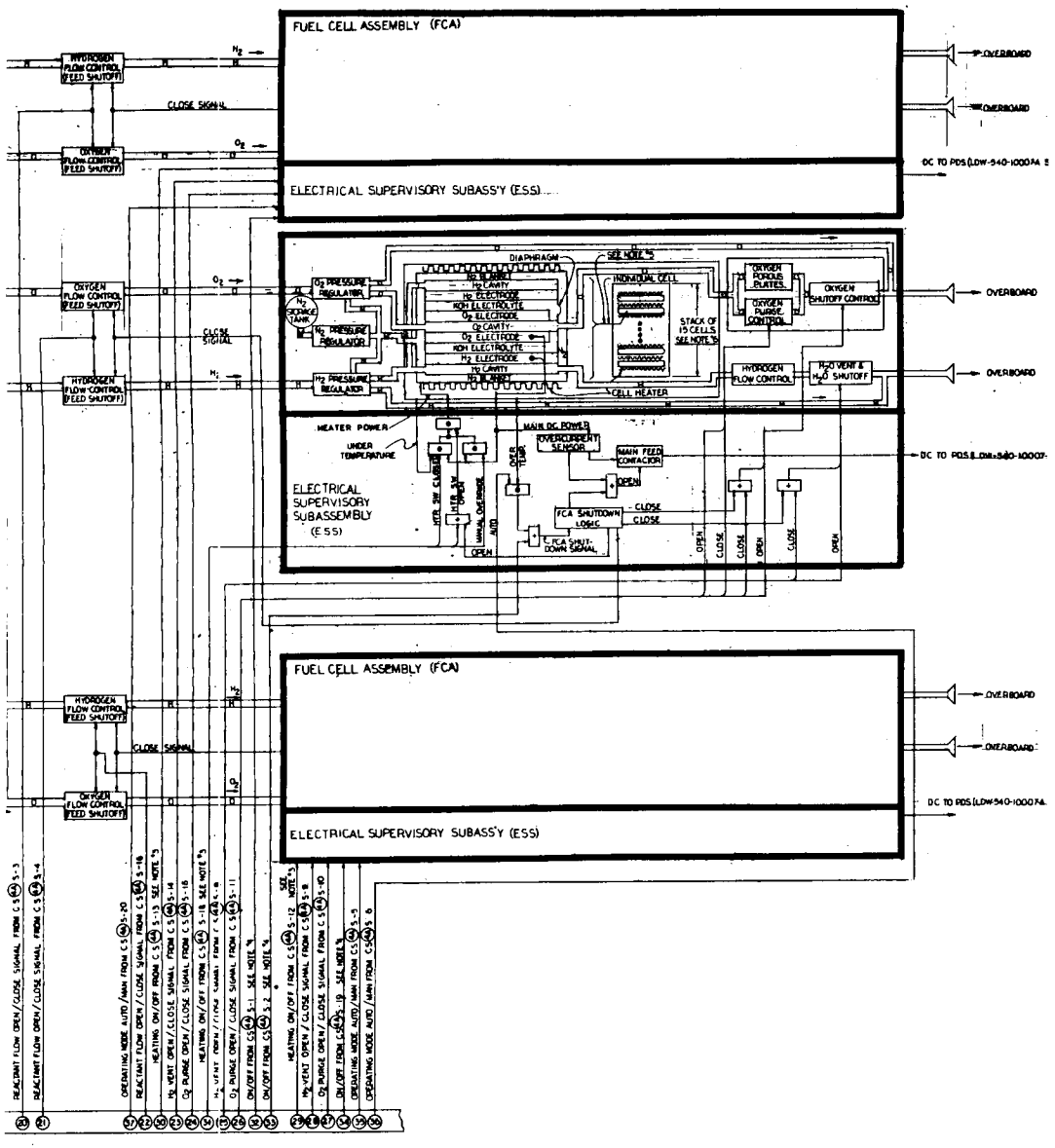
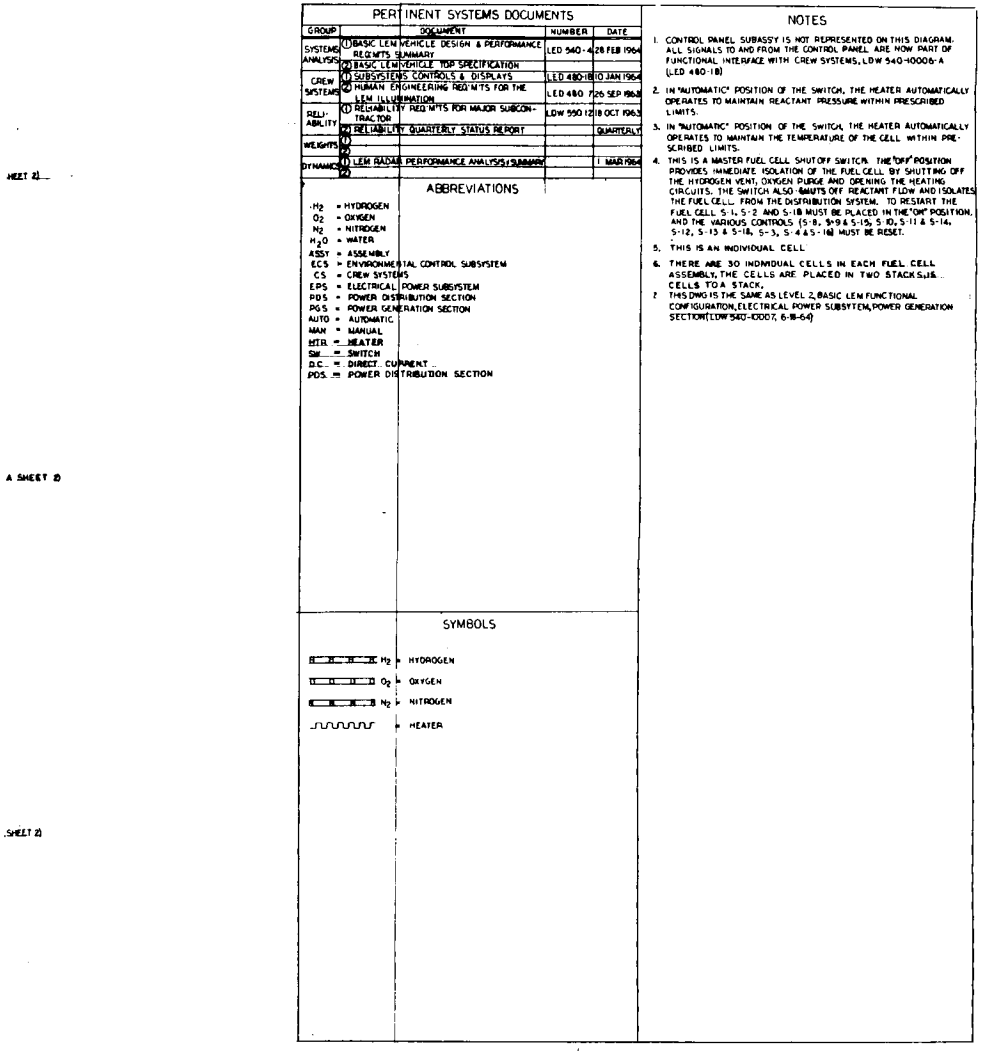


Figure 5-1 Functional Con

[redacted]



PERTINENT SYSTEMS DOCUMENTS

GROUP		DOCUMENT	NUMBER	DATE
SYSTEMS ANALYSIS	(1)	BASIC LEM VEHICLE DESIGN & PERFORMANCE REQ'MTS SUMMARY	LED 540-4	28 FEB 1964
	(2)	BASIC LEM VEHICLE TOP SPECIFICATION		
CREW SYSTEMS	(1)	SUBSYSTEMS CONTROLS & DISPLAYS	LED 480-18	10 JAN 1964
	(2)	HUMAN ENGINEERING REQ'MTS FOR THE LEM ILLUMINATION	LED 480 7	26 SEP 1963
RELIABILITY	(1)	RELIABILITY REQ'MTS FOR MAJOR SUBCONTRACTOR	LDW 550 12	18 OCT 1963
	(2)	RELIABILITY QUARTERLY STATUS REPORT		QUARTERLY
WEIGHTS	(1)			
	(2)			
DYNAMICS	(1)	LEM RADAR PERFORMANCE ANALYSIS SUMMARY		1 MAR 1964
	(2)			

ABBREVIATIONS

- H_2 = HYDROGEN
- O_2 = OXYGEN
- N_2 = NITROGEN
- H_2O = WATER
- ASSY = ASSEMBLY
- ECS = ENVIRONMENTAL CONTROL SUBSYSTEM
- CS = CREW SYSTEMS
- EPS = ELECTRICAL POWER SUBSYSTEM
- PDS = POWER DISTRIBUTION SECTION
- PGS = POWER GENERATION SECTION
- AUTO = AUTOMATIC
- MAN = MANUAL
- HTR = HEATER
- SW = SWITCH
- D.C. = DIRECT CURRENT
- PDS = POWER DISTRIBUTION SECTION

SYMBOLS

- H_2 = HYDROGEN
- O_2 = OXYGEN
- N_2 = NITROGEN
- = HEATER

NOTES

1. CONTROL PANEL SUBASSY IS NOT REPRESENTED ON THIS DIAGRAM. ALL SIGNALS TO AND FROM THE CONTROL PANEL ARE NOW PART OF FUNCTIONAL INTERFACE WITH CREW SYSTEMS, LDW 540-10006-A (LED 480-18)
2. IN "AUTOMATIC" POSITION OF THE SWITCH, THE HEATER AUTOMATICALLY OPERATES TO MAINTAIN REACTANT PRESSURE WITHIN PRESCRIBED LIMITS.
3. IN "AUTOMATIC" POSITION OF THE SWITCH, THE HEATER AUTOMATICALLY OPERATES TO MAINTAIN THE TEMPERATURE OF THE CELL WITHIN PRESCRIBED LIMITS.
4. THIS IS A MASTER FUEL CELL SHUTOFF SWITCH. THE "OFF" POSITION PROVIDES IMMEDIATE ISOLATION OF THE FUEL CELL BY SHUTTING OFF THE HYDROGEN VENT, OXYGEN PURGE AND OPENING THE HEATING CIRCUITS. THE SWITCH ALSO SHUTS OFF REACTANT FLOW AND ISOLATES THE FUEL CELL FROM THE DISTRIBUTION SYSTEM. TO RESTART THE FUEL CELL S-1, S-2 AND S-18 MUST BE PLACED IN THE "ON" POSITION, AND THE VARIOUS CONTROLS (S-8, S-9 & S-15; S-10, S-11 & S-14, S-12, S-13 & S-16, S-3, S-4 & S-16) MUST BE RESET.
5. THIS IS AN INDIVIDUAL CELL
6. THERE ARE 30 INDIVIDUAL CELLS IN EACH FUEL CELL ASSEMBLY, THE CELLS ARE PLACED IN TWO STACKS, 15 CELLS TO A STACK.
7. THIS DWG IS THE SAME AS LEVEL 2, BASIC LEM FUNCTIONAL CONFIGURATION, ELECTRICAL POWER SUBSYSTEM, POWER GENERATION SECTION (LDW 540-10007, 6-18-64)

A SHEET 2)

SHEET 2)

[redacted]

figuration, Electrical Power Subsystem, Power Generation Section

SUBSYSTEM: Electrical Power/Power Generation Section

5.2 WORKSHEETS

PHASE: Earth Launch to LEM Separation

EQUIPMENT/ FUNCTION	*FAILURE MODE*	*UNCOMPENSATED FAILURE EFFECT*	*MISSION COMPLETION MODE*	*MISSION ABORT MODE*	*REMARKS*
One of three fuel cell assemblies (FCA)	Over temperature	Fuel cell seal failure or flooding resulting in loss of fuel cell power.	Continue mission on two fuel cells. Over-temperature signal will automatically shut off failed FCA during unmanned phases. Monitor bus equipment will be dropped from bus system. Significance of bus loss discussed in Power Distribution Section (PDS).	Jettison LEM and return to primary earth landing site.	If FCA drops below "bootstrap" temperature after it is shut down, FCA cannot be reinstated. During manned phases, an attempt may be made to place FCA back on the line if temperature is not too low. Provision of warning light in CSM for unmanned phases being investigated.
	Reverse current	Cell flooding resulting in loss of fuel cell power	Continue missions on two fuel cells. Monitor bus equipment will be dropped from bus system.		Electrical supervisory subassembly (ESS) will open main line contactor. FCA will be placed in a standby mode and may be reinstated if fault is corrected. Essential and survival bus will provide power.
	Over current		Continue mission on two fuel cells. Monitor bus equipment will be dropped from bus system.		Main line contactor will open and FCA will be placed in a standby mode. FCA may be reinstated if fault is corrected.
Descent SH_2 tank	Total loss of SH_2 in failed tank	Reduced lunar stay time	Switch to ascent SH_2 tank. Loss of descent SH_2 tank at launch does not allow the LEM a lunar stay. Loss of descent SH_2 tank at lunar orbit checkout will allow a 5-hr lunar stay using ascent tanks (1).		The ΔP of O_2, H_2, and N_2 in FCA is very critical. Loss of correct ΔP will flood FCA and result in loss of power. PGS is investigating best mode of switching tanks without loss of pressure to FCA. A 75-min post-landing check plus 100-min prelaunch check has been assumed as ground rule for all lunar stays. Effect of a 5-hr lunar stay upon crew's normal rest and sleep cycle needs further study. (1) This precludes any subsequent orbital contingency. (2) The possibility of having the CSM provide SH_2, SO_2, or electrical power should be studied.
Descent SO_2 tank	Total loss of descent SO_2 in failed tank	Same as above	Loss of descent SO_2 tank at launch does NOT allow the LEM a lunar stay. Loss of descent SO_2 tank at end of lunar orbit checkout allows one man out for one hr, max.		ECS/EPS valve may be open prior to launch and will provide 7.1 lb additional reactant to FCAs, if descent tank fails. GOX tank will provide 16.7 hr of FCA operation at trans-lunar loads. Ascent tank will allow LEM to complete 55 additional hr of the mission at trans-lunar loads. GOX tank must be filled with high purity oxygen to be acceptable for FCA use. No orbital contingency is available for completion mode described.
One of two ascent SH_2 tanks	Total loss of ascent SH_2 reactant in one of two tanks	Reduced orbital contingency time	Continue mission using descent SH_2 tank plus remaining ascent tank for ascent.		No effect until ascent from lunar surface. A 3.5-hr orbital contingency is available. Ascent tank contains 6.91 lb of usable SH_2. Backup for lunar ascent SH_2 is lost.
Ascent SO_2 tank	Total loss of ascent SO_2 reactant in failed tank	Loss of redundant ascent oxygen	Continue mission using descent SO_2 tank plus GOX tank during ascent.		Backup to GOX tank and orbital contingency is lost during lunar ascent. There is no backup for descent SO_2 tank when cabin is pressurized from GOX.
GOX tank	Loss of GOX in failed tank	Loss of rapid cabin repressurization	Continue mission on descent SO_2 tank. Switch to ascent SO_2 tank at lunar launch.		Backup for ascent SO_2 tank is lost. Rapid pressurization of cabin is lost. ECS manual check valve must be in check position to prevent loss of descent tank at GOX failure. Orbital contingency can be 9 hrs. LEM can perform a normal reference mission lunar stay (24 hr).

SUBSYSTEM: Electrical Power/Power Generation Section

PHASE: LEM Separation to Landing

EQUIPMENT/ FUNCTION	*FAILURE MODE*	*UNCOMPENSATED FAILURE EFFECT*	*MISSION COMPLETION MODE*	*MISSION ABORT MODE*	*REMARKS*
One of three FCAs	Over temperature	Fuel cell seal rupture or flooding resulting in loss of fuel cell power	Continue mission on two fuel cells. FCAs are placed in manual mode upon entry into LEM. FCA must be shut down by System Engineer if over-temperature occurs.	Abort using essential and survival bus equipment. Staging will require switchover to ascent stage reactant tanks.	Essential and survival bus will provide power when one FCA is lost.
	Reverse current or over-current	Cell flooding resulting in loss of fuel cell power	Continue mission on two FCAs. See earth launch to LEM separation FCA data.		See earth launch to LEM separation FCA data.
Descent SH_2 tank	Total loss of descent SH_2 reactant in failed tank	Loss of orbital contingency; reduced lunar stay time	Switch to ascent SH_2 tanks.	Return to CSM using ascent SH_2 tanks.	Ascent tank heaters must be actuated before separation to bring ascent tanks up to an acceptable pressure level (approx. 200 psia). Ascent tanks must be able to supply pressure to FCAs without interruption. Continuous pressure prevents loss of FCAs due to flooding caused by loss of proper ΔP. Tank-switch method now under study. If tank is lost at separation, while FCAs are carrying normal electrical loads, LEM can complete post-landing checkout, 7-hr lunar stay, and prelaunch checkout. Orbital contingency and backup is lost. Failure at lunar landing results in a 10.5-hr stay.
Descent SO_2 tank	Total loss of descent SO_2 reactant in failed tank	Same as above	Switch to ascent SO_2 tank.	Switch to ascent SO_2 tank and return to CSM.	Failure at separation allows post-landing checkout, one-hr lunar exploration, and prelaunch checkout. LEM cabin remains unpressurized during the exploration. Failure of descent SO_2 tank at landing allows 3-hr lunar exploration (one man out) and pressurized cabin during ascent. No orbital contingency is available.
One of two ascent SH_2 tanks	Total loss of ascent SH_2 reactant in one of two tanks	Reduced orbital contingency time	Continue mission on descent SH_2 tank. Remaining ascent SH_2 tank will supply hydrogen during LEM ascent.	Aborts from 250 sec or less into powered descent may be made with descent stage; therefore, tankage will not require switching. If staging is required (i.e., for aborts from times greater than 250 sec), switch to remaining ascent SH_2 tank.	Backup on ascent to CSM is lost. A 3.5-hr orbital contingency is available.
Ascent SO_2 tank	Total loss of ascent SO_2 reactant in failed tank	No orbital contingency available in mission completion mode	Continue mission on descent SO_2 tank. GOX tank will supply oxygen to fuel cells during LEM ascent.	Aborts from 250 sec or less into powered descent may be made with descent stage; therefore, tankage will not require switching. If staging is required (i.e., for aborts from times greater than 250 sec), switch to GOX tank.	No backup O_2 available except PLSS. No orbital contingency.
GOX tank	Loss of GOX in failed tank	Loss of fast cabin repressurization capability, loss of ascent oxygen backup	Continue mission on descent SO_2 tank. Switch to ascent SO_2 tank at lunar launch.	Abort using descent SO_2 tank. If staging must be performed, switch to ascent SO_2 tank.	See remarks for GOX tank during Earth Launch to LEM separation phase.

SUBSYSTEM: Electrical Power/Power Generation Section

PHASE: Lunar Stay

EQUIPMENT/ FUNCTION	*FAILURE MODE*	*UNCOMPENSATED FAILURE EFFECT*	*MISSION COMPLETION MODE*	*MISSION ABORT MODE*	*REMARKS*
One of three FCAs	Over temperature	Fuel cell seal failure or flooding resulting in loss of fuel cell power	Continue mission on two fuel cells. FCAs are in manual mode. Failed FCA must be manually shut down if over-temperature occurs. See earth launch to LEM separation data.	Abort using essential, survival, and peak bus equipment.	Loss of one FCA requires reduction of power demand. Monitor bus will be dropped to prevent overloading remaining two FCAs. If FCAs are not operating at peak power, some monitor bus loads may be reinstated.
	Reverse current Over current	Cell flooding resulting in loss of fuel cell power	Continue mission on two FCAs. See earth launch to LEM separation data.		
Descent SO_2 tank	Total loss of descent SO_2 tank while following procedures are taking place: PLSS charging	Reduced orbital contingency time	Switch to ascent SO_2 tank. Open ECS/EPS valve to continue PLSS charging.	Same as for mission completion.	Prior to ascent, fill PLSS from ascent tank. A 9-hr orbital contingency is available if LEM makes an immediate ascent.
	GOX charging	Same as above	Switch to ascent SO_2 tank. Open ECS/EPS valve to continue GOX charging.	Same as for mission completion.	Fill GOX bottle from ascent tank prior to liftoff. This distributes available ascent oxygen between two tanks, thus reducing vulnerability to a second failure. During normal GOX charging (from descent SO_2), ECS/EPS valve should remain closed because, with this valve open, two orifices (acting as restrictors) will be placed in parallel. This will result in a lowering of SO_2 tank pressure because presently sized tank heaters are not sized to allow oxygen delivery at the two-orifice flow rate.
	Cabin repressurization	Same as above	Switch to ascent SO_2 tank. Open ECS/EPS valve for subsequent cabin repressurization.	Switch to ascent SO_2 tank. Open ECS/EPS valve for subsequent cabin repressurization prior to liftoff, or return to CSM in depressurized condition (hard-suit operation).	Ascent SO_2 tank will pressurize cabin in 51 min. Cabin requires 6.9 lb of oxygen for pressurization to 5.2 psia. Cabin can be pressurized from ascent SO_2 tank thus allowing GOX tank to remain filled. Approx. 3 hr will be available for orbital contingency after a 2-hr phasing wait.
Ascent SO_2 tank	Total loss of ascent SO_2 tank	Loss of orbital contingency time and backup oxygen supply	Continue mission on descent SO_2 tank until 10 min before lunar liftoff. Ascent must be performed with oxygen supplied from GOX.	Same as for mission completion.	PLSS and GOX must be filled before liftoff. No orbital contingency time is available; therefore, wait for proper CSM phasing before liftoff.
GOX tank	Total loss of GOX tank	Loss of backup oxygen supply during ascent to CSM	Continue mission on descent SO_2 tank until 10 min before lunar liftoff. Ascent must be performed with ascent SO_2.	Same as for mission completion.	If cabin in not pressurized at time of failure, an 8.5-hr orbital contingency is available. Fill cabin and PLSS from descent SO_2 tank prior to liftoff.
Descent SH_2 tank	Total loss of descent SH_2 tank	Reduced lunar stay time	Switch to ascent SH_2 tanks.	Switch to ascent SH_2 tanks and return to CSM.	Switching to ascent SH_2 tanks for a descent SH_2 tank failure at lunar landing allows post-landing checkout, 10.5-hr lunar stay, prelaunch checkout, and normal ascent to CSM. Backup is lost during ascent. Ascent SH_2 tanks will provide a 9-hr orbital contingency plus normal return to CSM if tanks are switched 100 min before ascent.
Ascent SH_2 tank	Total loss of one of two ascent SH_2 tanks	Loss of backup SH_2 during ascent	Complete normal lunar stay period using descent SH_2 tank. Ascend with remaining ascent SH_2 tank.	Abort using remaining ascent SH_2 tank.	A 3.5-hr orbital contingency is available if tank is placed in operation 100 min before launch.

SUBSYSTEM: Electrical Power/Power Generation Section

PHASE: Lunar Ascent

EQUIPMENT/ FUNCTION	*FAILURE MODE*	*UNCOMPENSATED FAILURE EFFECT*	*MISSION COMPLETION MODE*	*MISSION ABORT MODE*	*REMARKS*
One of three FCAs	Over temperature	Fuel cell seal failure or flooding resulting in loss of fuel cell power.	Continue mission on two fuel cells. FCAs are in manual' mode. Failed cell must be manually shut down if over-temperature occurs. See earth launch to LEM separation data.	Abort using essential, survival, and peak bus equipment.	
	Reverse current	Cell flooding resulting in loss of fuel cell power	Continue mission on two FCAs. See earth launch to LEM separation data.		
	Over current	Same as above	Same as above.		
Ascent SH_2 tank	Total loss of ascent SH_2 reactant in one of two tanks	Loss of backup SH_2 during ascent	Continue mission using remaining ascent SH_2 tank.	Not applicable.	A 3.5-hr orbital contingency is available if tank is placed in operation 100 min before launch. No backup available during ascent.
Ascent SO_2 tank	Total loss of ascent SO_2 reactant in failed tank	Loss of orbital contingency & backup oxygen supply	Continue mission using GOX tank.	Same as above.	No orbital contingency is available. GOX tank will provide 100 min of ECS operation on lunar surface plus direct ascent to CSM. FCA consumption for 90-min prelaunch checkout comes from residual in descent tank.
GOX	Total loss of GOX tank	Loss of backup oxygen supply	Continue mission using ascent SO_2 tank.	Same as above.	An 8.5-hr orbital contingency is available if an undelayed launch takes place and the cabin is left unpressurized. A 9-hr contingency is available if the cabin is pressurized before launch by descent SO_2 tank. The manual check valve must be in check position or ascent SO_2 tank is lost through GOX tank. PLSS may help to supply oxygen for orbital contingency requirements.
Cryogenic interstage disconnect	Failure of SO_2 or SH_2 disconnect to separate	Inability to ascend from lunar surface resulting in loss of crew.	None.	None.	

Electrical Power Subsystem/Power Generation Section (Cont.)

5.3 DESCRIPTION

The Electrical Power Subsystem (EPS) consists of a Power Generation Section (PGS) and a Power Distribution Section (PDS).

The EPS generates, distributes and controls the electrical power required for all phases of the LEM mission. In addition, the EPS provides oxygen to the Environmental Control Subsystem (ECS).

The PGS provides DC power to the PDS which in turn distributes power to the LEM vehicle. The PGS consists of subsections; Fuel Cell Subsection (FCS), Cryogenic Storage , and Supply Subsection (CSS).

5.3.1 Fuel Cell Subsection

The FCS consists of 3 Fuel Cell Assemblies (FCA) each providing 900 watts (nominal) power to the PDS at a voltage of 27 to 32 volts. The FCA operates on the principle of the Hydrox or Bacon type cell utilizing dual porosity nickel electrodes with an aqueous solution of potassium hydroxide (KOH) as electrolyte. The reactants combine in the cells at a temperature of 460 - 500^{o}F to produce power, water and waste heat. The waste heat and product water are exhausted overboard by allowing excess hydrogen to flow thru the fuel cells. This provides for a maximum reliability, minimum weight system.

The FCA consists of three major subassemblies:

- Fuel cell reactor subassembly (FCRS)
- Control Subassembly (CS)
- Electrical Supervisory Subassembly (ESS)

The FCRS consists of 30 individual cells connected electrically in series and assembled in two adjacent stacks. Manifolds distribute reactant gases from the regulators to each stack of cells and carry away the by-products. Cell heaters provide heat to the FCRS as required and electrically insulate adjacent cells. The two cell stacks are fastened to a common support structure which is attached to the LEM structure through four shock mounts.

The Control Subassembly performs the following functions:

- Controls pressure of the reactants, regulates heat and removes water.
- Provides storage and controls nitrogen pressurizing gas.
- Provides monitoring and protective service which ensures that the fuel cell operates within its design limitations.

The ESS monitors and controls the electrical operation as follows:

- Provides automatic shutdown and start up sequencing of the FCA.
- Provides overcurrent sensing and protection.

- Provides reverse current sensing and protection.
- Provides overtemperature protection circuits.

5.3.2 Cryogenic Supply Subsection (CSS)

The croygenic storage and supply tanks contain the hydrogen and oxygen necessary for operation of the fuel cells. The oxygen tanks also supplies the oxygen requirements of the Environmental Control Subsystem (ECS). The CSS consists of a hydrogen and oxygen tank in the descent stage. In addition, an oxygen and two hydrogen tanks are located in the aft equipment bay of the ascent stage. Tank pressurization is obtained by electrical heating elements located within the tanks and from ambient heat leak. A pressure switch automatically controls the amount of heat required to maintain the pressures required for single phase operation. The heaters operate in automatic mode with manual override capability. A pressure transducer, temperature sensor and capacitance type quantity gauge are included in all tank assemblies. Redundant electrical latching type shut-off valves are included with the tank assemblies in the ascent stage. An overpressure relief valve on each ascent tank is provided to vent excess tank pressure past the shut off valves into the flow line downstream. A series of lines, valves, heat exchanger, etc., carry oxygen and hydrogen from the tanks to the fuel cells as required during the mission.

5.3.3 Power Generation Operation

The cryogenic storage assemblies will be filled with the applicable fluids and capped at a predetermined time prior to earth launch. The flow of gaseous reactant for FCA start up will be provided by Ground Support Equipment. Switch over to internal reactant supply will be made before the Mobile Arming Tower moves away.

The hydrogen tank and the oxygen tank in the LEM descent stage contain the reactants to allow the FCA's to provide power to the LEM during Launch Countdown. Translunar, Lunar Descent and Lunar Stay time. In addition, oxygen required for the ECS is also contained in the oxygen storage tanks. Upon staging, the ascent oxygen tank and the two ascent stage hydrogen tanks provide the reactants for operating the ECS and each FCA. The GOX tank will provide redundancy for the ascent oxygen tank.

SECTION 6

ELECTRICAL POWER/POWER DISTRIBUTION SECTION

6.1 SUMMARY

The EPS Power Distribution Section contingency analysis can be summarized as follows:

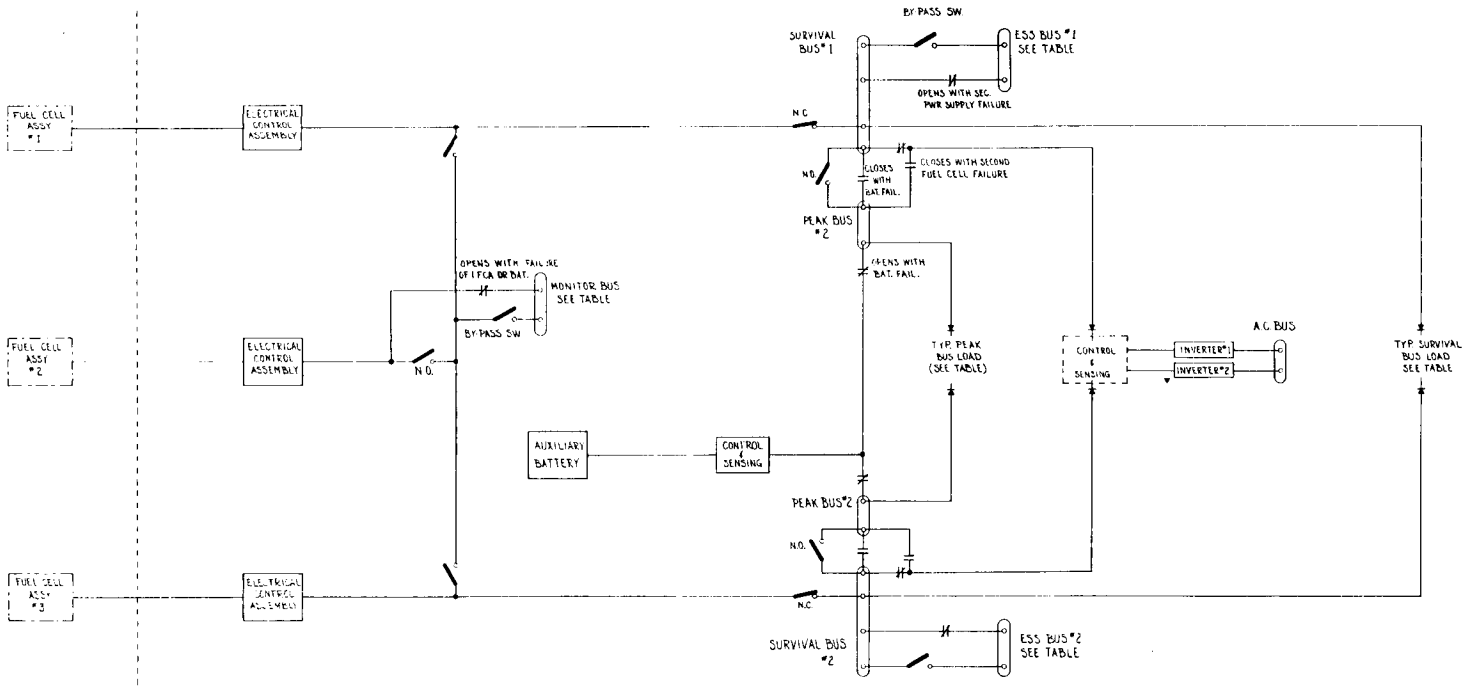
- No single failure, of either a power source or a bus, results in crew loss. Busses were assumed to have failed due to either an open circuit between the source and the bus, or a direct short circuit to the bus. Shorting of a bus is felt to be a remote possibility because of the insulation design, but is included for completeness.

- No single power source failure prevents completion of the mission.

- Single bus failures which prevent mission completion are:

  - Failure of survival bus #1 or essential bus #1 prior to touchdown prevents completion of the mission due to loss of the landing radar. A change is being considered at this time to crosstie essential #1 and #2 busses with survival #2 and #1 busses, respectively, via the bypass switches. This would provide completely redundant paths to the essential busses and remove the effect of loss of a single survival bus. These redundant circuits also make a power loss due to any open circuit between the survival and essential busses impossible for a single failure.

  - Failure of the AC bus due to shorting prior to touchdown prevents completion of the mission due to loss of gimbal drive actuators, RCS quantity gaging and altitude/altitude rate display. This failure is unlikely because of the redundant AC supply and low probability of loss by shorting. The remaining equipment (see work sheets) affected by loss of the AC bus is backed up by other equipment or modes of operation not dependent on AC power and therefore a mission completion mode is available.



LEM DISTRIBUTION FUNCTIONAL DIAGRAM
SHOWN IN NORMAL FLIGHT MODE

LEGEND:
- OPEN RELAY CONTACT
- CLOSED RELAY CONTACT
- MANUAL SWITCH

LOAD ARRANGMENT TABLE

SURVIVAL BUS
- NCG COMPUT.
- GDU
- FDAI
- ATCS (CES)
- RCS GAGING
- CMU
- DECA
- IMU
- TRANSPONDER
- CAUT & WARN
- TIMING EQUIPMENT
- FUEL CELL CONT
- GLYCOL PUMP
- DOCKING LT
- FLOOD LT
- VHF
- PMP
- AUDIO CENTERS

ESS #1 BUS
- LANDING RAD
- CABIN FAN

ESS #2 BUS
- ABORT GUIDANCE
- S-BAND

PEAK BUS
- DES ENGINE
- TCA (RCS)
- O2 HEATERS
- H2 HEATERS
- * INVERTER
- * SUIT FAN
- * REN. RAD

MONITOR BUS
- T.V. & T.V. LIGHT
- PCM
- S BAND AMP.
- OSCE
- INSTR.
- SCIENT INSTR.
- DSE

INVERTER
- E.L. LIGHTING
- GIMBAL DRIVE ACT.
- GAGING - EPS & RCS
- RENDEZVOUS RADAR
- DSE
- ALT/ALT RATE DISPLAYS
- AOT

* REN. RADAR, SUIT FAN, & INVERTER NORMALLY ON SURVIVAL BUS - SWITCHED TO PEAK BUS FOR 2 FUEL CELL FAILURE
** DOES NOT INCLUDE S-BAND POWER AMPLIFIER WHICH IS ON MONITOR BUS.
▼ NOT SHOWN ON M-5 MOCK-UP.

Figure 6-1 Functional Configuration,
Electrical Power Subsystem, Power Distribution Section

SUBSYSTEM: Electrical Power Subsystem/Power Distribution Section 6.2 WORKSHEETS *PHASE:* Earth Launch thru Final Docking

EQUIPMENT/ FUNCTION	*FAILURE MODE*	*UNCOMPENSATED FAILURE EFFECT*	*MISSION COMPLETION MODE*	*MISSION ABORT MODE*	*REMARKS*
One of three fuel cell assemblies (FCA)	No power output	Decreased power output to level where only essential and survival loads can be energized. For loads dropped, see monitor bus failure in this section	Continue with monitor bus off the line. In addition, VHF equipment must time-share with S-band equipment. Essential and survival equipments only are on the line. Use essential load profile.	Same as mission completion mode. Return to CSM. Jettison LEM and abort to primary earth landing site.	Monitor bus can be put back on the line manually with a bypass switch but other loads must be turned off selectively (e.g., cabin fan off, S-band power amplifier and PCM on). During lunar surface operations, normal requirements can be handled by the two remaining FCAs. The battery is normally off the line during lunar stay. Loss of S-band implies loss of telemetry but telemetry is dropped with monitor bus.
Two of three FCAs	No power output	Decreased power output to level where only survival loads can be energized. Essential loads dropped: cabin fan, abort guidance, landing radar, S-band emergency voice.	• Lunar orbit checkout to touchdown: none (1) • Touchdown to liftoff: Continue with limited lunar operations (2). • Liftoff to final docking: use survival equipment and survival power profile.	Return to CSM using survival equipment and survival power profile. Jettison LEM and abort to primary earth landing site. Same as above. Same as mission completion mode.	(1) Suit fan, rendezvous radar, and inverter are automatically switched to battery to reduce load to within FCA limits. Total energy demands on battery exceed capacity. Battery is sized to handle lunar descent requirements with two FCAs operative, plus ascent requirements with one FCA operative (essential profile during descent, survival profile during ascent). (2) Careful monitoring of FCA voltage, current and temperature required to prevent exceeding its power rating (900 watts). Battery is kept off line until lunar liftoff. Survival load requirements during lunar stay are approximately 550 watts, average. This assumes cabin fan off and IMU and computer in standby. Loads not exceeding 350 watts can be turned on selectively (e.g., cabin fan plus S-band power amplifier plus PCM alternatively, PLSS battery charger can be used in place of these equipments but not concurrently with them).
All three FCAs	No power output	Decreased power output making survival impossible in some phases	None.	• Separation to touchdown: very marginal; reduce loads to absolute minimum and return to CSM. • Lunar stay thru power ascent: none. • Coast thru dock: return to CSM with absolute minimum loading.	Battery not sized to handle situation (See above). There is only enough energy to handle survival loads for a very short duration.
Battery	No power output	Momentary loss of peak bus equipment: descent engine control valves, RCS TCA's, O_2 heaters, H_2 heaters	Monitor bus is automatically dropped and FCA power automatically transferred to peak bus. Resulting equipment status is same as for case of one FCA failure, i.e., continue with essential load profile.	Same as mission completion mode. Return to CSM. Jettison LEM and abort to primary earth landing site.	Monitor bus may be put back on line with the bypass switch but other loads must be turned off selectively (e.g., cabin fan off, PCM and S-band power amplifier on). Battery is on line during LEM flight and checkout phases only.
Inverter	No power output	Loss of redundant AC power source	Continue with remaining inverter	Same as mission completion mode. Return to CSM. Jettison LEM and abort to primary landing site.	

SUBSYSTEM: Electrical Power Subsystem/Power Distribution Section *PHASE:* Earth Launch thru Final Docking

EQUIPMENT/ FUNCTION	*FAILURE MODE*	*UNCOMPENSATED FAILURE EFFECT*	*MISSION COMPLETION MODE*	*MISSION ABORT MODE*	*REMARKS*
Battery charger	No power output	Cannot recharge PLSS batteries	For failure prior to first lunar exploration: Preferred mode is to limit lunar explorations to one and perform normal ascent with fully charged spare battery. Alternate mode is to perform the design reference mission explorations and ascend with uncharged PLSS. For failure between first lunar exploration and end of lunar stay: variable number of recharges of the one PLSS can be achieved.	Return to primary earth landing site.	Present PLSS design provides one spare rechargeable battery for the two PLSS carried aboard LEM. The choice between 3 explorations and uncharged PLSS during ascent versus one exploration and fully charged PLSS during ascent involves a tradeoff between partial mission success and crew safety.

SUBSYSTEM: Electrical Power Subsystem/Power Distribution Section *PHASE:* Earth Launch thru Final Docking

EQUIPMENT/ FUNCTION	*FAILURE MODE*	*UNCOMPENSATED FAILURE EFFECT*	*MISSION COMPLETION MODE*	*MISSION ABORT MODE*	*REMARKS*
AC bus	No power output	Loss of AC loads: altitude/range display, altitude rate/range rate display, EPS quantity gaging, gimbal drive actuator (GDA), electroluminescent panel lighting, AOT light, data storage equipment (DSE), RCS quantity gaging	• Powered descent to touchdown: none, due to loss of altitude information and GDAs (1). • Touchdown to liftoff: launch time computed by CSM using its RR and LEM transponder. • Liftoff to final docking: midcourse and terminal rendezvous performed by LEM using CSM RR and LEM transponder (2).	Return to CSM using LEM transponder and CSM rendezvous radar (2). Jettison LEM and abort to primary earth landing site. Same as mission completion mode. Same as mission completion mode.	(1) Loss of GDAs may be offset by use of RCS for c-g control during powered descent. But RCS quantity gaging is also lost and additional RCS propellant required makes it mandatory that RCS quantity be known. RCS penalties resulting from GDA loss are discussed in S & C Subsystem worksheets. (2) Midcourse and terminal rendezvous manouvers to be performed by LEM on instructions transmitted by CSM via VHF. CSM performs V required and time-to-burn computations. This mode requires that both CSM and LEM IMUs have same reference.
Essential bus No. 1 and survival bus No. 1	No power output	Essential bus No. 1: loss of abort guidance and landing radar: Survival bus No. 1: above loss plus removal of redundant path to survival loads	• Prior to touchdown: None. Cannot perform powered descent without landing radar. • Failure of survival bus No. 1 after touchdown: continue by using remaining path to survival loads. • Failure of essential bus No. 1 after touchdown: continue by using primary guidance	Return to CSM using survival loads. Jettison LEM and abort to primary earth landing site.	It may be possible to correct this failure by manually closing bypass switch. If fault is corrected, a normal mission completion is possible but automatic bus dropout feature is lost. Consequently, closer crew monitoring of fuel cell status is required.
Essential (1) bus No. 2 and survival bus No. 2	No power output	Essential bus No. 2: loss of cabin fan and S-band transponder assembly (3) survival bus No. 2: above loss plus removal of redundant path to survival loads	Failure of essential bus No. 2: close faceplate and operate ARS loop in closed loop mode(2). Communication restricted to VHF operation during LOS between LEM and CSM (2). Failure of survival bus No. 2: continue by using remaining path to survival loads but without essential bus No. 2 (2, 3).	Same as mission completion mode. Return to CSM. Jettison LEM and abort to primary earth landing site.	(1) It may be possible to correct this failure by manually closing bypass switch. If fault is corrected, a normal mission completion is possible but automatic bus dropout feature is lost. Consequently, closer crew monitoring of fuel cell status is required. (2) Loss of cabin fan does not negate ability to maintain pressurized cabin. Therefore, suits remain soft except during periods of cabin decompression (EVA operations). (3) Loss of S-band results in loss of telemetry. Telemetry is presently considered essential to the mission, Recent communications subsystem changes provide for a second S-band transponder assembly. Power source for the second transponder assembly is under study.
Peak bus No. 1 or 2	No power output	Loss of redundant path to peak loads	Continue by using remaining path.	Same as mission completion mode. Return to CSM. Jettison LEM and abort to primary earth landing site.	
Monitor bus	No power output	Loss of the following monitor bus loads: S-band power amplifier (3), instrumentation (signal conditioners and sensors), PCM, DSE, on board checkout, ECS cabin and suit pressure and temperature gages, ECS glycol pressure and temperature gages, RCS pressure and temperature gages, PLSS battery charger, TV, TV light, and scientific instrumentation	Continue with essential equipment. (1). Loss of PLSS battery charger restricts lunar exploration. (2).	Same as mission completion mode. Return to CSM. Jettison LEM and abort to primary earth landing site.	(1) An attempt to restore bus may be made by manually closing bypass switch. This overrides automatic dropout switch. Caution must be taken not to overload FCA and cause voltage to drop below permissible limits. (2) See battery charger failure for description. (3) Loss of S-band power amplifier reduces operational capability of this equipment to emergency voice only. Telemetry is presently considered essential to the mission, consequently the S-band, PCM and instrumentation equipment should not be dropped. Recent communications subsystem changes provide for a second S-band transponder assembly. Power source for the second transponder assembly is under study.

Electrical Power Subsystem/Power Distribution Section (Cont.)

6.3 DESCRIPTION

The Electrical Power Subsystem/Power Distribution Section (EPS/PDS) provides electrical power for all subsystems in the LEM except the pyro devices (see Figure 6-1). The sources of power are three fuel cell assemblies (part of PGS) and an auxiliary battery (part of PDS) which operate at a nominal 28 VDC. The fuel cell assemblies are connected in parallel and energize the two survival busses, the two essential busses and the monitor bus. The battery energizes the two peak busses. Two redundant inverters convert the DC to single phase AC at 115V, 400 cps, and energize the AC bus.

Automatic switching is provided to keep the loading consistent with the capacity of the operating power supplies in the event of a power failure. If any fuel cell assembly fails, the monitor bus is de-energized and the two remaining assemblies energize the essential and survival busses. If a second fuel cell assembly fails, the essential busses are also de-energized and the remaining fuel cell assembly shares the survival bus loads with the battery. If the battery fails, the monitor bus is de-energized and the peak busses are transferred to the fuel cell assemblies. If a fuel cell assembly fails in addition to the battery the essential busses are also de-energized and the two remaining fuel cell assemblies share the peak and survival bus loads.

In all the above cases the de-energized busses may be re-energized via a manual bypass switch but the current and voltage must be carefully monitored to prevent overloading.

SECTION 7

ENVIRONMENTAL CONTROL SUBSYSTEM

7.1 SUMMARY

7.1.1 The results of the contingency analysis performed on the Environmental Control Subsystem indicate that failure of the ECS heat transport section prevents the performance of a safe abort during any phase of the mission, due to the loss of thermal control of the cryogenic oxygen and hydrogen which then leads to failure of the fuel cells. Provisions for emergency heating are being studied.

The analysis further indicates that the following failures would require abort of the mission:

- Failure of the oxygen supply section relief valve in the open position results in leakage of the descent stage SOX at a rate of 7 1/2 lbs. per hour. At this leak rate, lunar exploration cannot be achieved if this failure occurs prior to the end of lunar orbit checkout. For failure after lunar orbit checkout, a limited lunar stay is possible. This assumes, however, that the steady leakage of oxygen from the failed tank does not constitute a hazard to other LEM functions.
- Failure of the descent stage water tank prior to lunar landing can result in loss of the IMU by overheating, since no water is supplied to the glycol loop water boiler.
- Failure of the glycol loop prior to lunar landing.

7.1.2 The contingency analysis has brought to light one area which will require further study and coordination with other LEM subsystems. The Power Distribution Section bus configuration shows the glycol pumps and suit circuit fans on the survival bus. The automatic switching controls for these components and the cabin pressure sensors, however, are supplied by the essential bus. If the LEM were required to perform an abort with survival equipments only, it would be desirable to have the automatic controls for these equipments available. Automatic switching of these equipments is not absolutely essential since manual switching can be accomplished. However, since these switching devices require relatively little power, the possibility of powering them from the survival bus should be studied.

7.1.3 In the previous draft of this report, it was assumed that all equipment powered by the survival bus should also be supplied by the emergency cooling loop. It is now realized that the above assumption, while generally correct, does not hold in all cases. One example stated that since guidance is essential for aborts, one would assume that PGNS should receive both emergency power and cooling. There are design problems, however, in providing PGNS with emergency cooling, and so AGS is used for aborts when primary cooling is lost. This is therefore one case in which PGNS is not supplied with both emergency power and cooling, but the abort guidance capability is retained for both primary cooling and power failures. Other apparent inconsistencies cited no longer exist because of configuration and ground rule changes.

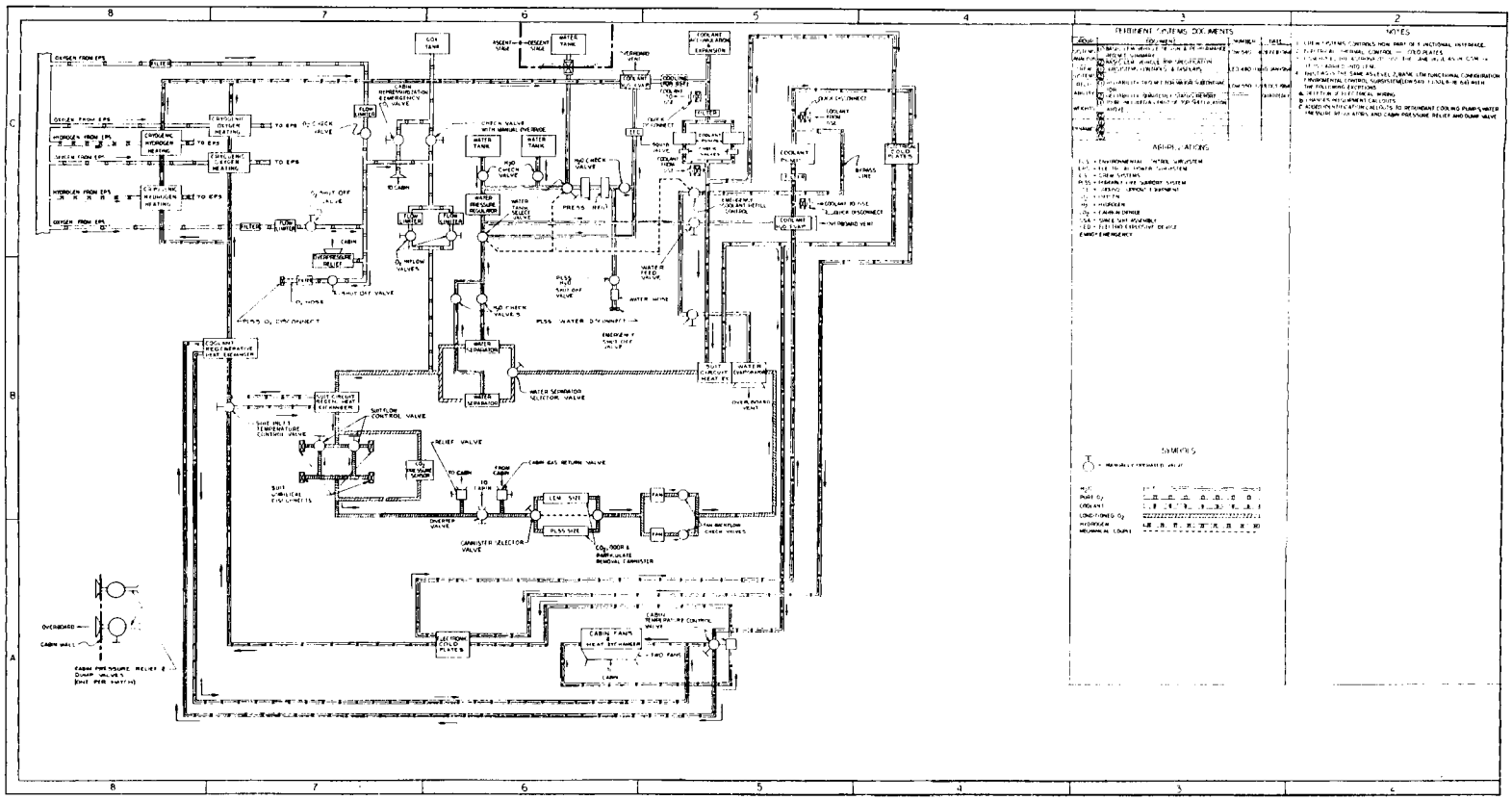


Figure 7-1 Functional Configuration, Environmental Control Subsystem

SUBSYSTEM: Environmental Control 7.2 WORKSHEETS *PHASE:* Earth Liftoff Thru Final Docking

EQUIPMENT/ FUNCTION	*FAILURE MODE*	*UNCOMPENSATED FAILURE EFFECT*	*MISSION COMPLETION MODE*	*MISSION ABORT MODE*	*REMARKS*
Water Management Section, Heat Transport Section	Primary water pressure regulator closed, squib valve doesn't fire at earth launch, check valve downstream of pressure regulator closed, coolant water evaporator failed, plugged line or loss of glycol	Failure of cold-plate-cooled equipment; no temperature or humidity control in ARS, no thermal control of cryogenics	None	Prior to separation: Jettison LEM. Separation to docking: None.	Failure of the heat transport section negates cryogenic thermal control which progressively leads to failure of the fuel cells. Electrical heating of the cryogenics as a backup is being studied. If this backup is implemented and the emergency cooling loop is required, third glycol pump must be manually selected, and shut-off valves to suit emergency water boiler and backup boiler must be manually opened. Cabin temperature control is not possible. The equipment presently on emergency cooling loop is: Rate Gyro Assembly; Abort Sensor Assembly; Abort Electronic Assembly; Flight Director Attitude Indicator; Rendezvous Radar Transponder; Rendezvous Radar Electronics; Attitude and Translation Assembly; VHF; Electrical Supervisory Subsystem; Signal Conditioners Electronic Assembly; Pulse Code Modulation Timing Equipment; Inverter; and S-Band.
Water Management Section	Loss of descent water tank, interstage disconnect closed	Same as above	Prior to lunar landing: To be studied. Lunar stay: Switch to ascent tanks and perform minimum mission.		
	Loss of one ascent water tank	Loss of ascent redundancy and orbital contingency capability	Normal	Normal	
	Inability to recharge PLSS due to water hose rupture, disconnect failed closed, shut-off valve failed closed	PLSS limited to initial charge	Perform minimum mission. Ascend with uncharged PLSS.	Normal	Assumed that two PLSSs are carried and initially charged by CSM during translunar flight.
	Water separator check valve closed	No humidity control in ARS	Manually switch to redundant separator.	Same as mission completion mode.	
	Glycol pump failure	Loss of thermal control	Redundant pump will automatically switch in.	Same as mission completion mode.	

SUBSYSTEM: Environmental Control

PHASE: All Manned LEM Phases

EQUIPMENT/ FUNCTION	*FAILURE MODE*	*UNCOMPENSATED FAILURE EFFECT*	*MISSION COMPLETION MODE*	*MISSION ABORT MODE*	*REMARKS*
Oxygen Supply Section	Emergency flood valve fails to open electrically.	Lose capability for fast cabin repressurization.	Lunar orbit checkout to LEM jettison - manually control flood valve (1).	Same as for mission completion mode. Return to CSM. Jettison LEM and abort to primary earth landing site.	(1) Loss of automatic operation of flood valve makes LEM cabin vulnerable to pressure drop due to micrometeorite puncture. Close monitor of cabin pressure would be required.
	Emergency flood valve fails to close after having been opened electrically.	Loss of GOX(2) and loss of cabin puncture protection.			(2) Depletion of GOX tank through open valve occurs in approximately 2 minutes.
	Fwd. cabin hatch, pressure relief valve fails open.	Lose O_2 from LEM.	Earth launch to transposition docking - normal continue (1).	Jettison LEM and abort to primary earth landing site.	(1) Failure in this phase requires no corrective action until transposition docking.
		Lose O_2 from CM.	Transposition docking - blow down CM, perform hard dock, enter LEM using PLSS, manually close valve, return to and repressurize CM(2).	Same as above	(2) To perform manual corrective action at this time is optional. Manual closure of fwd. valve could be delayed until LEM occupancy at lunar orbit checkout.
		Loss of GOX and cabin pressure.	Lunar orbit checkout to LEM jettison - manually close valve (3).	Same as above	(3) After cabin pressure drops to 4.7 psia (due to failed open relief valve) flood valve will be automatically initiated to check cabin decay. However, warning system is initiated with flood valve open. Therefore, this failure cannot go undetected.
	Upper cabin hatch, pressure relief valve fails open.	Requires one additional LEM pressurization from CM.	Transposition docking - repressurize LEM from CM and perform normal hard dock.	Same as above	
GOX tank	Loss of GOX due to tank leakage.	Loss of capability for fast cabin repressurization.	See EPS/PGS worksheets.	See EPS/PGS worksheets	
	O_2 relief valve fails open.	Loss of descent SOX(1)	Earth launch to lunar checkout - none (2). Lunar orbit checkout to lunar liftoff - perform limited lunar exploration (3).	Return to CSM. Jettison Lem and abort	(1) Failure would also result in loss of GOX if the overridable check valve were in check position. For this reason, it is recommended that this valve be normally closed. (2) Relief valve failure results in loss of descent SOX at 7.5 lbs/hr initial leak rate. Insufficient descent SOX is available to perform even a minimum lunar landing mission. (3) Amount of survival time to perform lunar exploration depends upon where in the phase the failure takes place.
ARS - Gas circulation.	No flow from suit fan #1, fan backflow check valve #1 closed or fan backflow check valve #2 open.	Pressurized phases: Loss of cooling, and water removal from suit Unpressurized phases: Loss of breathing oxygen, suit cooling and water removal	Suit fan #2 is automatically switched in.	Same as for mission completion mode. Return to CSM. Jettison LEM & abort to primary earth landing site.	

SUBSYSTEM: Environmental Control

PHASE: All Manned LEM Phases

EQUIPMENT/ FUNCTION	*FAILURE MODE*	*UNCOMPENSATED FAILURE EFFECT*	*MISSION COMPLETION MODE*	*MISSION ABORT MODE*	*REMARKS*
ARS-Gas circulation	Either suit flow control valve failed closed.	Physiological deterioration of either crewman due to lack of body cooling within approximately 20 minutes.	Appropriate astronaut disconnects hose from failed valve & reconnects to free suit connector on other astronauts suit(1).	Same as for mission completion mode. Return to CSM. Jettison LEM and abort to primary earth landing site.	(1) Suit cooling capacity with buddy system is presently being studied. Operation in this mode may require a reduction in crew activity level to cope with the reduced suit cooling capacity.
	Puncture of any component within the suit loop.	Reduced gas conditioning depending upon the size of the puncture.	During pressurized cabin phases - normal continue (1).	Same as above	(1) Since the ARS equipments are located entirely within the LEM pressurized area, no gas is lost from punctures or leaks.
		Loss of suit loop pressure resulting in death of crew.	During unpressurized cabin phases - use PLSS (2).	Repressurize cabin. Return to CSM & abort to primary earth landing site.	(2) Oxygen will be lost during the time that the cabin is decompressed.
ARS - humidity and temp. control.	One of two water separator's fails.	No moisture removal from atmosphere resulting in crew discomfort due to high humidity and reduced cooling.	Manually switch to redundant water separator.	Same as for mission completion mode.	For average metabolic rates, a 235 Ft^3 Cabin & atmosphere initially dry, it will take approximately 36.5 minutes to attain 100% relative humidity at $75^{o}F$. Entrained water would enter the space suits following failure of the water separator.
	No glycol flow to suit H/X due to plugged line or temp. control valve failed closed.	Crew discomfort due to cool gas entering suit.	Reduce gas flow to suit via suit flow control valve. Increase cabin temp. by manually controlling cabin temp. control valve.		Amount of discomfort depends on metabolic load. Suit inlet temp. varies from approximately 47^{o} F 66^{o} F with no glycol flow thru suit H/X. Affect is simultaneous for both astronauts.
	Suit temp. control valve fails open.	Crew discomfort due to warm gas entering suit (1).	Increase gas flow to suit via suit flow control valve. Decrease cabin temp. by manually overriding cabin temp. control valve.		(1) With this failure, suit inlet temp. could reach 95^{o} F without corrective action.
ARS - gas purification	Mechanical failure of LiOH cartridge.	Incomplete CO_2 removal resulting in crew asphyxiation and/or unpleasant odors.	Manually switch to bypass cartridge. Remove and replace defective cartridge.		
	Canister selector valve mechanically jammed.	Cannot switch to alternate cartridge, cannot replace used cartridge.	Blow down cabin to expel CO_2 saturated gas. Recompress and continue mission.		For average metabolic rates, CO_2 builds up within LEM at the rate of 5.8 mmhg per hour. Crew can tolerate approximately 25 mmhg for 3 hrs. It takes approximately 4 hours to build up CO_2 to a pressure of 25 mm after recycling the cabin gas.

Environmental Control Subsystem (Cont.)

7.3 DESCRIPTION

The Environmental Control Subsystem (ECS) consists of five integrated sections: atmosphere revitalization, oxygen supply and cabin pressure control, heat transport, water management, and cold plate (see Figure 7-1). The major portion of the ECS equipment is in the pressurized compartment of the ascent stage. A glycol loop and a gaseous oxygen accumulator are in the ascent stage equipment bay section. Two water tanks are in the tankage section of the ascent stage, while a third (larger) water tank is in the descent stage.

The ECS controls the oxygen in, pressurization of, ventilation of, and temperature of the cabin and the space suits worn by the two astronauts. It provides breathing oxygen for the astronaut's space suits and cabin; it limits the level of carbon dioxide and removes odors, moisture, and particulate matter from the oxygen breathed by the crew; and it automatically controls the temperature of the electronic equipment. The ECS also stores water for drinking, food preparation, and the Portable Life Support System (PLSS).

7.3.1 Atmosphere Revitalization Section

The atmosphere revitalization section consists of a suit circuit assembly and cabin recirculation assembly which conditions the oxygen for cooling and ventilating the space suits worn by the crew and monitors cabin oxygen recirculation and temperature. More specifically, the suit circuit assembly controls the carbon dioxide level of the atmosphere breathed by the crew, removes odors and noxious gases from the

crew atmosphere, removes foreign objects within the space craft, removes excess moisture from the cabin atmosphere, and provides control of the space suit gas temperature and the gas flow through the space suits.

The suit circuit assembly consists of two (one redundant) suit circuit fans, a suit circuit heat exchanger and water evaporator, and two (one redundant) water separators. In addition, the section contains a regenerative heat exchanger, a carbon dioxide partial pressure sensor, two carbon dioxide odor removal canisters, relief valve, check valves and interconnecting tubing.

Oxygen from the oxygen supply and cabin pressure control section is circulated through the suit circuit assembly by one of the two redundant suit circuit fans. As either of the fans can maintain the required suit-circuit oxygen flow, only one fan is operated at a time. After leaving the fan, the oxygen passes through the suit circuit heat exchanger, which transfers excess heat from the oxygen to the heat transport section coolant. The suit circuit water evaporator removes the excess heat in the event of failure of the suit circuit heat exchanger. Moisture that condenses when the oxygen passes through the suit circuit heat exchanger or suit circuit water evaporator is removed from the oxygen by one of the two redundant water separators. Each separator can meet the water removal requirements but depending on the position of the manually operated water separator selector valve, only one of the separators will function at a time.

Downstream from the two water separators is the suit circuit regenerative heat exchanger, which allows the temperature of the oxygen to be manually controlled by the crewmember before it enters the space suits. Warm coolant from the heat transport section flows through the heat exchanger, transferring heat to the oxygen. The temperature of the oxygen is controlled by varying the flow of coolant through the heat exchanger; the astronaut manually controls this flow by the suit temperature control valve.

In addition, the astronauts control their comfort by their individual flow control valve. The carbon dioxide partial pressure sensor monitors the carbon dioxide partial pressure level. The oxygen then passes through one of the two redundant carbon dioxide and odor removal canisters, each consisting of a canister and replaceable cartridge, and once again flows into one of the two suit circuit fans where the cycle is repeated.

During open-faceplate operation (normal pressurization level), the suit-circuit diverter valve is opened to pass the entire oxygen flow from the suit circuit assembly into the cabin. This ensures that a sufficient amount of cabin oxygen is circulated through the suit circuit assembly to maintain the desired carbon dioxide and humidity levels in the cabin. In the event of a decompression of the cabin atmosphere, the cabin pressure switch provides a signal that automatically closes the diverter valve. The suit-circuit relief valve prevents over-pressurization of the assembly. When the pressure is 4.4 psia or greater, the relief valve is fully open; when the pressure is less than 4.1 psi, the relief valve is fully closed.

Recirculation and temperature control of the cabin oxygen is provided by the cabin recirculation assembly. The assembly contains fans that recirculate the oxygen and a cabin heat exchanger that automatically heats or cools the oxygen. Heat is transferred between the cabin gas and heat transport section coolant that flows through the cabin heat exchanger. The temperature of the coolant is controlled in the heat transport section.

7.3.2 Oxygen Supply and Cabin Pressure Control Section.

The oxygen supply and cabin pressure control section provides oxygen required by the atmosphere revitalization section and also supplies oxygen to refill the PLSS, used during exploration of the lunar surface. In addition, the section maintains cabin pressure by supplying oxygen at a rate equal to cabin leakage plus crew consumption, allows for cabin depressurization and subsequent pressurization by the astronauts and maintains space suit pressure during unpressurized cabin operation.

The oxygen supply and cabin pressure control section consists of a gaseous oxygen accumulator, two (one redundant) oxygen demand regulators, various check valves, shutoff valves and interconnecting tubing.

At the normal pressurization level, the pressure in the cabin and the space suits is maintained at 5±0.2 psia, which permits the astronauts to open their faceplates and remove their gloves. With the cabin depressurized, the space suits must be sealed and the pressure in the suits is reduced to the egress mode level of $3.7\ ^{+.02}_{-.00}$ psia.

The oxygen stored in the supercritical storage assembly is sufficient for the required cabin repressurizations and refills of the PLSS primary oxygen storage tanks, in addition to normal crew consumption and vehicle and space suit leakage. The gaseous oxygen accumulator is used in conjunction with normal oxygen flow for cabin repressurizations that require high oxygen flow rates. The self-sealing PLSS oxygen disconnect permits refilling the PLSS primary oxygen storage tanks.

Pure oxygen from the supercritical storage assembly in the descent stage passes through one of the cryogenic oxygen heat exchangers where it is warmed by the heat transport section coolant to make it compatible with the operation of components in the oxygen supply and cabin pressure control section and the atmosphere revitalization section. Two oxygen demand regulators (one redundant), each with a manual override control the delivery of oxygen to the atmosphere revitalization section in response to signals from pressure sensors. The cabin repressurization and emergency oxygen valve monitors the delivery of oxygen to the cabin for cabin repressurization or to slow the loss of cabin pressure in the event of punctures in the cabin pressure shell, in response to signals from the cabin pressure switch. In addition, the valve has a manual override.

Overpressurization of the oxygen supply and cabin pressure control section is prevented by the oxygen pressure relief valve, which automatically relieves excess pressure by venting oxygen into the cabin. Overpressurization of the cabin is prevented by the cabin pressure relief and dump valves which automatically relieves excess cabin pressure by venting oxygen overboard or can be manually operated, from inside or outside the cabin, to dump the cabin pressure overboard.

7.3.3 Heat Transport Section.

The heat transport section of the LEM consists of two closed-loop systems - the primary and the secondary. Each of these systems circulates an ethylene glycol-water coolant to provide temperature control of the electronic equipment. In addition, the primary system provides temperature control of the oxygen circulated through the cabin and the space suits and warms the cyrogenically stored oxygen and hydrogen.

The primary coolant loop consists of two coolant pumps (one redundant), the cabin temperature control valve, coolant regenerative heat exchanger, Freon boiler, coolant water evaporator, coolant accumulator, coolant filter, various valves, and interconnecting tubing. The secondary coolant loop, used for cooling of critical equipment in the event of primary system failure, consists of a coolant pump, coolant water evaporator, filter, valves, and interconnecting tubing.

In the primary coolant loop, the coolant is circulated by one of the two coolant pumps. Each pump can provide normal flow and only one pump is operated at a time. After leaving the pump, the coolant flow divides, some going through the suit-circuit heat exchanger to transfer the heat from the oxygen to the coolant and some going through part of the cold plate section where it absorbs heat from the electronic equipment. The flow then divides between the regenerative heat exchanger and its bypass obtaining the required heat for the cabin heat exchanger from the regenerative heat exchanger.

The flow split is made by the cabin temperature control valve which is controlled by the cabin heat exchanger glycol-water discharge temperature. The cabin heat exchanger discharge coolant temperature is maintained within a narrow range of temperature which in turn maintains the heat exchanger discharge temperature and cabin temperature within the required limits.

The coolant then passes through another parallel cold plate section where it receives waste heat from the electronic equipment. After passing the cold plate section, the flow is controlled to the suit circuit regenerative heat exchanger for suit heating by the suit temperature control valve. Warm coolant then flows in parallel paths through the cryogenic oxygen and hydrogen heat exchangers to warm the oxygen and hydrogen supplied from the supercritical storage assemblies. Waste heat is removed from the

coolant in the water evaporator by a sublimation process and its products discharged overboard. The coolant then flows through the coolant filter (which removes particles that could cause a malfunction) and into the coolant pumps continuing the cycle. The coolant accumulator maintains pressure above the coolant vapor pressure in the heat transport section and accommodates volumetric changes of the coolant.

Ground support provisions are provided for by two sets of two self sealing quick disconnects for the primary and secondary circuits. Each set of fittings provide for supply and return of the GSE coolant and are used for coolant fill and draining and ground cooling during equipment and system checkouts.

The secondary circuit coolant pump which only operates during flight if the primary system fails, circulates coolant through the emergency equipment - cold plates, water boiler and filter. The water boiler removes heat from the coolant by evaporation.

7.3.4 Water Management Section.

The water management section of the ECS stores the water for the metabolic needs of the crew, vehicle cooling and, for refilling the PLSS water tanks. The water management section consists of two water tanks in the ascent stage, one in the descent stage, water pressure regulators, check valves, shutoff valves and interconnecting tubing.

The water tanks are pressurized prior to earth launch to maintain the required pumping pressure in the water tanks. The water tank in the descent stage supplies a major portion of the water required up to the time of lunar launch. After lunar launch, water is obtained from the two smaller tanks in the ascent stage. In addition to water from the tanks, water from the atmosphere revitalization section water separators is used in

the water management section. The self-sealing PLSS water disconnect permits filling the PLSS water tanks and delivering of metabolic water for drinking and food preparation.

7.3.5 Cold Plate Section

A cold plate is thermally affixed to each piece of electronic equipment requiring active temperature control. Coolant from the heat transport section passes through the cold plates and removes waste heat from the electronic equipment. There are two basic types of cold plates - structural and non-structural. Structural cold plates provide mounting support for the electronics as well as being a heat sink. Non-structural cold plates provide only a heat sink and are installed between the electronics and its supporting structure. Cold plates which serve electronics required for abort contain two separate coolant flow passages - one for primary loop coolant, the other for secondary loop coolant. All other cold plates contain only a single flow passage for primary loop coolant.

SECTION 8

STRUCTURAL/MECHANICAL SUBSYSTEM

8.1 SUMMARY

If no capability for extra-vehicular operations (EVO) or extra-vehicular transfer (EVT) were implemented, failures in the Structural/Mechanical Subsystem which would cause loss of the crew are:

- The drogue "probe head securing fingers" do not capture the probe during rendezvous docking.
- The upper hatch (X Axis) cannot be opened for ingress to the CM after rendezvous docking.
- The drogue cannot be removed from the tunnel after rendezvous docking.
- The landing gear fails during lunar touchdown in a manner which precludes ascent.

If capability for EVT is implemented, the only single failure considered which would result in loss of the crew is the landing gear failure.

With respect to mission completion. the following single failures make even a minimum lunar landing mission impossible if EVT capability is not implemented:

- The drogue cannot be removed from the LEM tunnel for crew transfer at time of lunar orbit checkout.
- The upper hatch (X axis) cannot be opened for crew transfer at time of lunar orbit checkout.
- The landing gear does not deploy or lock in extended position during checkout in lunar orbit.
- The drogue "probe head securing fingers" do not capture the probe during transposition docking or when the drogue is reinstalled in the tunnel prior to lunar orbit separation.
- Loss of vision thru both "Z" axis windows prior to lunar landing.

- The forward hatch (Z axis) cannot be opened for egress to lunar surface.
- Loss of thermal shielding in quantities and locations which make thermal control impossible.

With EVT capability, mission completion is possible despite the first two failures and probably possible despite the third.

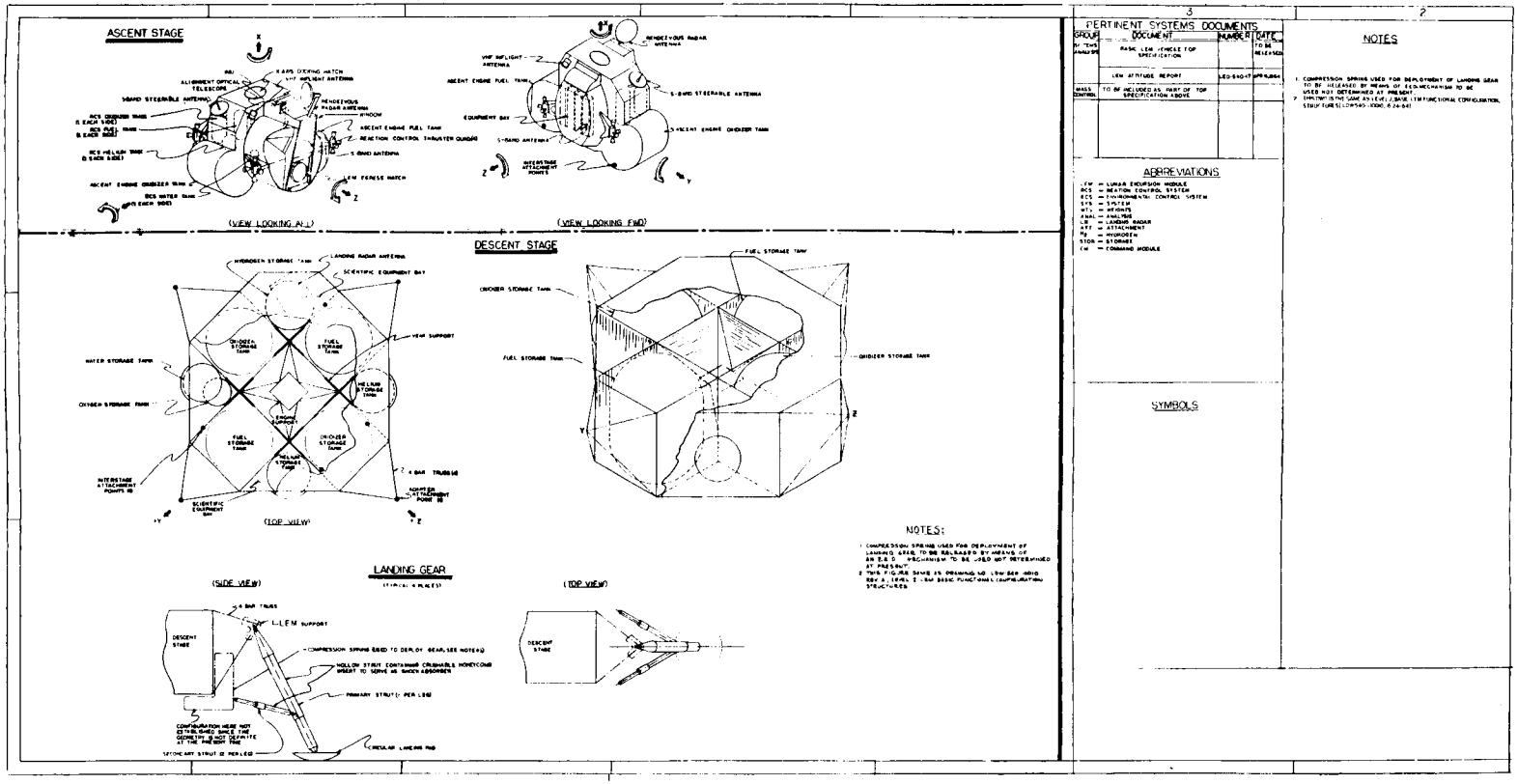
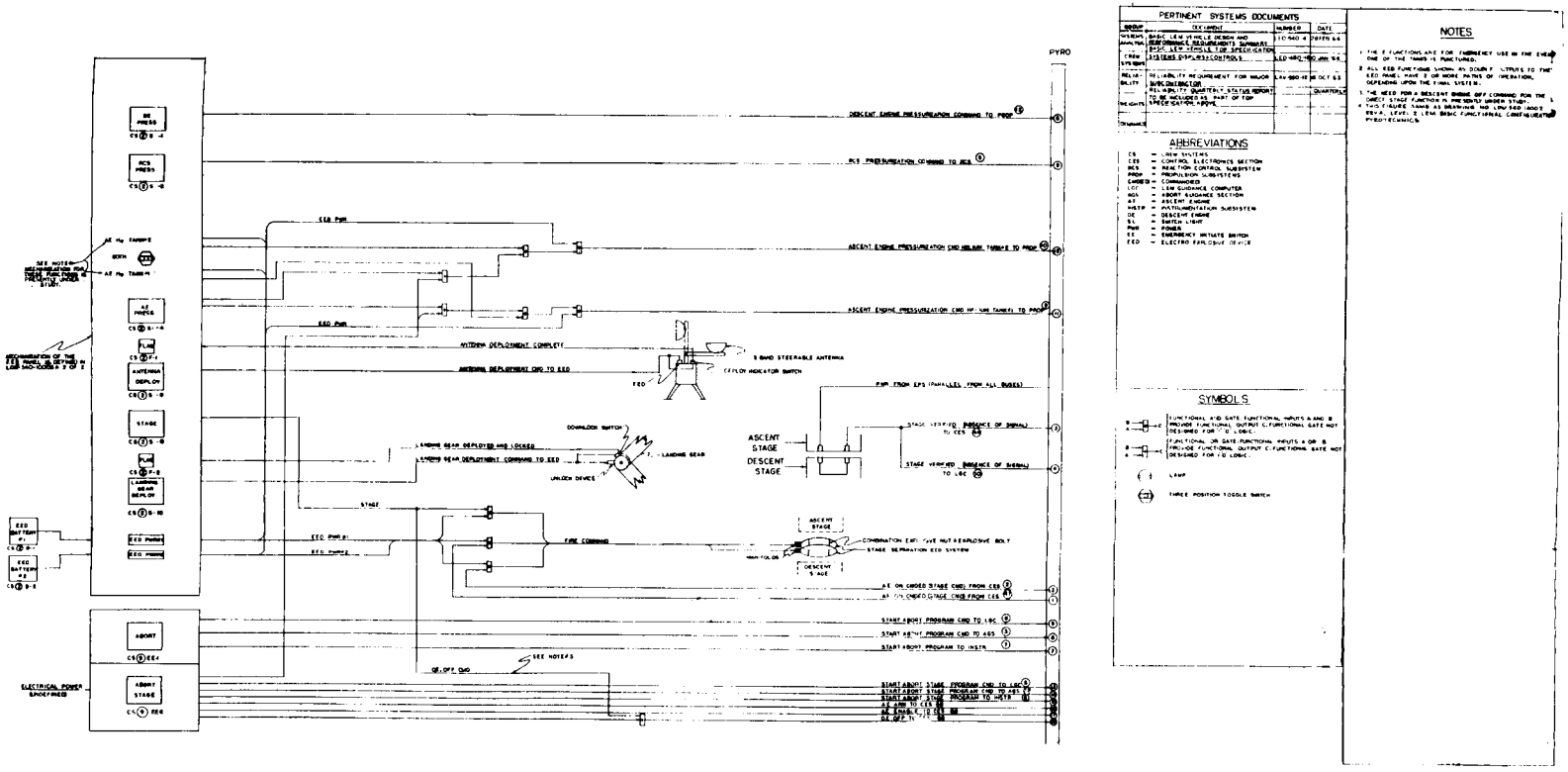


Figure 8-1 Functional Configuration, Structures


PYRO
DESCENT ENGINE PRESSURIZATION COMMAND TO PROP
RCS PRESSURIZATION COMMAND TO RCS
ASCENT ENGINE PRESSURIZATION CMD HELIUM TANKS TO PROP
ASCENT ENGINE PRESSURIZATION CMD HP GAS TANKS TO PROP
S BAND STEERABLE ANTENNA
ANTENNA DEPLOYMENT COMPLETE
ANTENNA DEPLOYMENT CMD TO EED
EXPLOY INDICATOR SWITCH
PWR FROM EPS (PARALLEL FROM ALL BUSES)
DOWNLOCK SWITCH
LANDING GEAR
LANDING GEAR DEPLOYED AND LOCKED
LANDING GEAR DEPLOYMENT COMMAND TO EED
UNLOCK DEVICE
ASCENT STAGE
DESCENT STAGE
STAGE VERIFIED (PRESENCE OF SIGNAL) TO CES
STAGE VERIFIED (ABSENCE OF SIGNAL) TO LGC
STAGE
EED PWR
FIRE COMMAND
COMBINATION EXPLOSIVE NUT & EXPLOSIVE BOLT
STAGE SEPARATION EED SYSTEM
AE ON ORBED STAGE CMD FROM CES
AE ON ORBED STAGE CMD FROM CES
START ABORT PROGRAM CMD TO LGC
START ABORT PROGRAM CMD TO AGS
START ABORT PROGRAM TO INSTR
SEE NOTE 5
GE, OFF CMD
START ABORT STAGE PROGRAM CMD TO LGC
START ABORT STAGE PROGRAM CMD TO AGS
START ABORT STAGE PROGRAM TO INSTR
AE ARM TO CES
AE ENABLE TO CES
DE OFF TO
ELECTRICAL POWER SPECIFIED
ABORT
ABORT STAGE
EED BATTERY #1
EED BATTERY #2
EED PWR #1
EED PWR #2
SEE NOTE 3
ABBREVIATIONS
CS = LEM SYSTEMS
CES = CONTROL ELECTRONICS SECTION
RCS = REACTION CONTROL SUBSYSTEM
PROP = PROPULSION SUBSYSTEMS
CMDED = COMMANDED
LGC = LEM GUIDANCE COMPUTER
AGS = ABORT GUIDANCE SECTION
AE = ASCENT ENGINE
INSTR = INSTRUMENTATION SUBSYSTEM
DE = DESCENT ENGINE
SL = SWITCH LIGHT
PWR = POWER
EI = EMERGENCY INITIATE BUTTON
EED = ELECTRO EXPLOSIVE DEVICE
SYMBOLS
FUNCTIONAL AND GATE FUNCTIONAL INPUTS A AND B PROVIDE FUNCTIONAL OUTPUT C FUNCTIONAL GATE NOT DESIGNED FOR T D LOGIC
FUNCTIONAL OR GATE FUNCTIONAL INPUTS A OR B PROVIDE FUNCTIONAL OUTPUT C FUNCTIONAL GATE NOT DESIGNED FOR T D LOGIC
LAMP
THREE POSITION TOGGLE SWITCH
PERTINENT SYSTEMS DOCUMENTS
DOCUMENT
NUMBER
DATE
NOTES

[redacted]

Figure 8-2 Functional Configuration, Pyrotechnics

SUBSYSTEM: Structural/Mechanical 8.2 WORKSHEETS *PHASE:* Transposition Docking

EQUIPMENT/ FUNCTION	*FAILURE MODE*	*UNCOMPENSATED FAILURE EFFECT*	*MISSION COMPLETION MODE*	*MISSION ABORT MODE*	*REMARKS*
Probe	Strut jams Collet fingers inoperative	Unable to secure soft dock	None.	Jettison LEM and S-IVB and abort to primary earth landing site.	No soft dock precludes hard dock. Probe and drogue have been joined together.
Drogue	Probe head securing fingers inoperative	Same as above.	None	Abort to primary earth landing site.	No soft dock precludes hard dock. LEM and S-IVB have not been secured to CSM. CM will jettison probe and latches prior to entry.
Interface seal	Damaged	Unable to hold 3.5 psia in interlock	Monitor pressure in CM & interlock, pressurize suits, blow down CM, hard dock, close CM hatch, repressurize.	Close valve in CM hatch, jettison LEM, abort to primary earth landing site.	CM hatch cannot be opened until pressure is equal. Hatch must be opened for hard dock and connection of umbilical.
Hard latches	Inoperative	Unable to secure hard latch	None.	Jettison LEM and abort to primary earth landing site.	If all latches (12) are not required, minimum number necessary for SPS firing during midcourse correction and LOI should be determined for structural integrity.
Cabin skin	Ruptured (during adapter separation)	Loss of cabin pressure	All LEM operations in hard-suit mode.	Same as above.	Assuming failure does not cause loss of structural integrity.
Ascent stage thermal shielding (cabin)	Damaged (during adapter separation)	Cabin heat load increase	Normal to extent ECS can support temperature increase.	Same as above.	If extensive damage occurs to thermal shielding depressurize LEM and operate in hard suit. Vehicle can be oriented to reduce solar heat input if damage is known.
Ascent stage thermal shielding (equip. bay)	Same as above	Equip. bay heat load increase	Same as above.	Same as above.	Glycol loop heat load increase will require additional water for cooling.
Ascent stage thermal shielding (prop. tanks)	Same as above	Same as above	Same as above.	Same as above.	Same as above.
Descent stage thermal shielding	Same as above	Tank compartments heat load increase	Rotate vehicle so damaged area is away from sun.	Same as above.	Relief valves will prevent excessive pressure in prop. tanks. Knowledge of lost insulation would allow vehicle positioning to reduce solar heating on this area.
Electrical receptacle	Pins damaged	Cannot make intervehicle umbilical connection.	None.	Same as above.	If umbilical from CSM cannot be engaged with receptacles in LEM, jettison of S-IVB cannot be accomplished.

SUBSYSTEM: Structural/Mechanical

PHASE: Lunar Orbit Coast Prior to LEM Checkout

EQUIPMENT/ FUNCTION	*FAILURE MODE*	*UNCOMPENSATED FAILURE EFFECT*	*MISSION COMPLETION MODE*	*MISSION ABORT MODE*	*REMARKS*
Probe	Release mechanism inoperative	Crew cannot transfer into LEM	None.	Replace ablative and pressure hatches. Jettison LEM and abort to primary earth landing site.	Probe cannot be removed from tunnel. No EVA from CM is possible with Block 1 CSM when docked to LEM.
Drogue	Probe head securing fingers inoperative (won't release)	Crew cannot transfer into LEM	None.	Same as above.	Neither probe nor drogue can be removed from tunnel. Now EVA from CM is possible with Block 1 CSM when docked to LEM.
Upper hatch	Relief and dump valve jammed	Unable to equalize pressure in interlock and LEM	Open hatch sufficiently to equalize pressure before opening completely (leak rate controlled by initial part of latch release cycle).	Jettison LEM and abort to primary earth landing site.	Hatch translates into LEM prior to rotation. A + P on CM side will escape into LEM during this translation.
Upper hatch	Locking mechanism jammed closed	Unable to open hatch	None.	Place probe and drogue in LEM tunnel, replace ablative and pressure hatches, jettison LEM and abort to primary earth landing site.	Cannot transfer crew into LEM. No EVA from CM is possible with Block 1 CSM when docked to LEM.
Cabin skin	Ruptured (meteoroids)	Loss of cabin pressure	All LEM operations in hard-suit mode	Same as above	Skin rupture would cause loss of pressure only and not affect structural integrity. Structure can operate in zero-pressure environment.

SUBSYSTEM: Structural/Mechanical

PHASE: Lunar Orbit Checkout to Separation

EQUIPMENT/ FUNCTION	*FAILURE MODE*	*UNCOMPENSATED FAILURE EFFECT*	*MISSION COMPLETION MODE*	*MISSION ABORT MODE*	*REMARKS*
Upper hatch	Locking mechanism jammed open	Cannot lock hatch	Allow hatch to remain open.	Jettison LEM and return to primary earth landing site.	Hatch will remain in place during zero-g. After separation, pressure will hold hatch in position.
Probe	Supporting mechanism inoperative	Probe cannot be secured in CM tunnel	None.	Place probe and drogue in LEM, transfer crew, install hatches in CM tunnel, jettison LEM and abort to primary earth landing site.	Without probe and drogue in place, soft dock, which is required for release of latches, cannot be achieved.
	Collet fingers inoperative	Cannot produce soft dock	None.	Same as above.	Same as above.
Drogue	Probe head securing fingers inoperative	Cannot produce soft dock	None.	Same as above.	Same as above.
	Supporting mechanism inoperative	Cannot be secured in LEM tunnel	None.	Same as above.	Same as above.
Hard latches	Inoperative	Cannot separate vehicles	None.	Same as above.	Twelve latches are engaged. Failure to release of any one would prevent vehicles from separating.
System Engineer's outer window	Crazed (meteoroids)	Loss of vision thru window	All operations requiring external visibility assumed by Commander.	Jettison LEM and return to primary earth landing site.	Loss of one window greatly reduces external vision required for best choice of landing site. Additional yaw maneuvering required to achieve total site survey.
Commander's outer forward window	Same as above	Same as above	All operations requiring external visibility assumed by Systems Engineer.	Same as above.	Same as above.
Both forward outer windows	Same as above	Complete loss of fwd external vision	None.	Place probe and drogue in LEM, transfer crew, install hatches in CM tunnel, jettison LEM and abort to primary earth landing site.	Would not be able to choose landing site.
Pyro battery	Internal short which drops output voltage	Propulsion system and RCS pressurization squib valves do not fire	Alternate circuit using second battery fires squibs.	Jettison LEM and return to primary earth landing site.	Have two completely independent circuits simultaneously initiated by one switch.
	Internal short which drops output voltage	Cannot extend landing gear	Alternate circuit using second battery fires unit.	Same as above.	There are two independent pyrotechnic circuits, each having its own battery and inititiated simultaneously by one switch.
Landing gear	Pyrotechnic device does not fire; gear does not deploy	Cannot make lunar landing	None.	Place probe and drogue in LEM, transfer crew, install hatches in CM tunnel, jettison LEM and abort to primary earth landing site.	Two initiators at each leg provide release for deployment. One initiator would do the job. The initiators are in separate circuits. With EVA from LEM crewman could manually (with tools) release gear if foot and hand grips are provided.
	Won't lock in extended position	Cannot make lunar landing	None.	Same as above.	With EVA from LEM crewman could manually activate gear if foot and hand grips are provided.
	Won't extend	Cannot make lunar landing	None.	Same as above.	Same as above.
S-band steerable Antenna	Won't deploy	Cannot communicate with MSFN	Use CM S-band via hardline intercom for voice.	Jettison LEM and abort to primary earth landing site.	With EVA from LEM crewman could manually deploy antenna. No telemetry available from LEM until lunar landing, via erectable S-band antenna.
Cabin skin	Ruptured (meteoroids)	Loss of cabin pressure	All LEM operations in hard-suit mode.	Same as above.	Skin rupture would cause loss of pressure only and not affect structural integrity. Structure can operate in zero-pressure environment.

SUBSYSTEM: Structural/Mechanical

PHASE: LEM Separation to Landing

EQUIPMENT/ FUNCTION	*FAILURE MODE*	*UNCOMPENSATED FAILURE EFFECT*	*MISSION COMPLETION MODE*	*MISSION ABORT MODE*	*REMARKS*
Commander's forward outer window	Crazed (meteoroids)	Loss of vision thru window	All operations requiring external visibility assumed by Systems Engineer.	Return to CSM. Jettison LEM and return to primary earth landing site.	Additional yaw maneuvering required to achieve total site survey.
Systems Engineer's outer window	Same as above	Loss of vision thru window	All operations requiring external visibility assumed by Commander.	Same as above.	Same as above.
Both outer windows	Same as above	Complete loss of external vision along Z axis	None.	Same as above.	Would not be able to choose landing site.
Cabin skin	Ruptured (meteoroids)	Loss of cabin pressure	All LEM operations in hard-suit mode.	Same as above.	Skin rupture would cause loss of pressure only and not affect structural integrity. Structure can operate in zero-pressure environment.
Equip. bay black box support structure	Deformation due to hi-g landing causing separation of equip. from cold plate	Decrease in heat sink area	Normal, to extent equipment can operate with elevated ambient temperature.	Same as above.	As glycol temperature would be reduced due to a decrease in heat input, some means of measuring internal black box temperature may be in order.
Upper hatch seal	Damaged	Cannot hold minimum pressure in LEM	All LEM operations in hard-suit mode.	Same as above.	Damaged when hatch was closed; pressure loss not detected until separation.
Pyro separation bolts	Premature ignition	Stages separate prematurely	None.	Initiate normal separation procedure and return to lunar orbit for redocking with CSM.	If failure occurred at touchdown, continue mission in normal fashion.
Landing gear	Strut buckles at impact	LEM pitches over	None.	None.	Possible use of RCS to give corrective pitching moment.

SUBSYSTEM: Structural/Mechanical *PHASE:* Lunar Stay

EQUIPMENT/ FUNCTION	*FAILURE MODE*	*UNCOMPENSATED FAILURE EFFECT*	*MISSION COMPLETION MODE*	*MISSION ABORT MODE*	*REMARKS*
Cabin skin	Ruptured (meteoroids)	Loss of cabin pressure	All LEM operations in hard-suit mode.	Normal, except crew operates in hard suit.	Skin rupture would cause loss of pressure only and not affect structural integrity. Structure can operate in zero-pressure environment.
Forward hatch	Hatch lock mechanism jammed closed	Cannot use fwd hatch for egress/ingress	None.	Stage and follow normal ascent mode.	No provisions for descending from LEM upper surface to lunar surface if upper hatch is used.
Upper hatch	Same as above	Cannot use tunnel area for lunar surface survey	Use windows for limited field of vision.	Same as above.	Later use at rendezvous docking requires EVA.
Upper hatch	Relief and dump valve jammed	Cannot depressurize	Use valve in fwd hatch.		LEM hatches cannot be opened with internal positive pressure.
Forward hatch	Same as above	Same as above	Use valve in upper hatch.		Same as above.
Commander's forward outer window	Crazed (meteoroids)	Loss of vision thru window	Onboard crewman uses Systems Engineer's window for visual check on outside crewman.		Due to structure on fwd face, outside view area greatly reduced by loss of one window.
Systems Engineer's outer window	Same as above	Same as above	Same as above (using Commander's window).		Same as above.
Both forward outer windows	Same as above	Complete loss of vision along Z axis	With depressurized LEM, use upper tunnel as a vantage point for external observations.		Considered a single failure.
Ascent stage thermal shielding (aft equip. bay)	Damaged	He tank in direct sunlight	None.		Loss of He thru pressure relief valve would reduce amount available to pressurize propellant tanks for successful ascent.
Ascent stage thermal shielding (prop. tanks)	Damaged	Tank compartment heat load increase	None.		Relief valves will prevent excessive pressure in prop. tanks.
Ascent stage thermal shielding (cabin)	Damaged	Cabin heat load increase	Normal to the extent that ECS can support temperature increase.		

SUBSYSTEM: Structural/Mechanical

PHASE: Lunar Launch to Crew Transfer

EQUIPMENT/ FUNCTION	*FAILURE MODE*	*UNCOMPENSATED FAILURE EFFECT*	*MISSION COMPLETION MODE*	*MISSION ABORT MODE*	*REMARKS*
Probe	Strut jams Collet fingers inoperative	Unable to secure soft dock	Extravehicular crew and equipment transfer thru LEM forward hatch and CM docking tunnel after disconnecting probe from drogue.	Same as mission completion mode.	LEM forward hatch can be used for EVT after removal of hatch. CM and LEM must be depressurized. LEM must back away from CM; requires tethering lines.
Drogue	Probe head securing fingers inoperative	Unable to secure soft dock	Same as above.		Same as above.
Drogue	Supporting mechanism inoperative	Cannot be removed from tunnel	Disconnect probe and separate vehicles. Transfer thru LEM forward hatch and via space environment to CM (crew and equipment).		CM and LEM must be depressurized.
Hard latches	Inoperative	Unable to secure hard dock	Same as above.		Present design has 12 latches taking midcourse correction loads. However, number of latches required to be engaged can be reduced; this number to be determined.
Interface seal	Damaged	Unable to hold pressure in interlock	Operate in pressurized suit mode.		Requires depressurization of CM during period of crew and equipment transfer.
Upper or forward hatch seal	Damaged	Unable to hold 3.5 psia cabin pressure	Same as above.		Same as above.
Ascent-Descent umbilical	Pyrotechnics (cable doesn't shear)	Ascent stage re-straining force at liftoff	None.	None.	This would be a double failure as pyrotechnic system is a redundant one having two independent devices and circuits.
Pyro battery	Internal short which drops output voltage	Unable to supply power for pyro-technic-operated disconnect	Alternate circuit using second battery fires unit.	Same as for mission completion mode.	Redundant system used. If both batteries are "out" it would be necessary for crewman to manually disengage umbilical and bolts holding ascent and descent stages together (double failure).
Commander's forward outer window	Crazed (meteoroids)	Loss of vision thru window	All operations requiring external vision assumed by Systems Engineer.		
Systems Engineer's forward outer window	Same as above	Same as above	All operations requiring external vision assumed by Commander.		
Both forward outer windows	Same as above	Complete loss of forward vision	Liftoff to docking accomplished in automatic mode. Window over commander used for docking maneuvers.		In backup guidance mode, all sightings would be made thru upper window.
Commander's upper outer window	Same as above	Loss of vision thru window (X axis)	Liftoff to docking accomplished in automatic mode. CM active during docking phase.		In backup guidance mode, all sightings would be made thru forward windows.
Interstage pyro devices (explosive bolts and nuts)	Do not fire	Cannot stage for ascent	Crewman manually unbolts ascent from descent stage.		Unbolting operation would depend on attitude of vehicle on lunar surface. If completely disjoined, ascent stage may slide to unfavorable position for launch (double failure).
Cabin skin	Ruptured (meteoroids)	Loss of cabin pressure	All LEM operations in hard suit mode.		Skin rupture would cause loss of pressure only and not affect structural integrity. Structure can operate in zero pressure environment.

Structural/Mechanical Subsystem (Cont.)

8.3 DESCRIPTION

The LEM structure is designed in two sections, ascent stage and descent stage (see Figure 8-1) with an interstage connection joining the two.

8.3.1 Ascent Stage

The ascent stage structure supports a cabin capable of operating either pressurized or unpressurized as required by the mission. The cabin holds all supplies required by the crew during its use and provides support for all controls and displays. Its atmosphere is controlled by the ECS. Windows located in the forward and upper cabin surface provide outside vision for the crewmen during the landing and rendezvous-docking operations. Two hatches are available for cabin ingress and egress. One hatch is located in the area of the docking structure for intravehicular transfer when the LEM is docked to the C/SM. The other is placed on the front face for use when transferring crew and equipment to and from the lunar surface and when the LEM is on the pad prior to launch.

Structural members support the propulsion system and provide correct positioning of the RCS thrust chamber assemblies relative to the body axes and C.G. location. Guidance and navigation equipment requiring close-tolerance alignment relative to dependent equipment is mounted on accurately positioned rigid members to ensure necessary alignment. An unpressurized area is provided with structural supports to accommodate the primary electrical power supplies (fuel cells) and equipment not requiring in-flight crew access. The supporting structure for this equipment also serves as a heat transfer link by permitting the flow of glycol through it.

Concentric with the X axis (which is also the nominal center line of thrust of the ascent and descent engines) at the top of the ascent stage is a ring which provides a structural interface for joining the LEM to the CSM. It is compatible with the clamping mechanisms housed in the CM and ensures structural continuity for transmitting midcourse correction and lunar orbit injection loads. The drogue portion of the docking mechanism is secured below this ring during the docking operation to mate with the CM mounted probe, and when "out of crew compartment" storage is required.

Supported externally by the ascent stage structure are the various antennas used by the Guidance and Communications Subsystems. Due to the space limitations within the S-IVB adapter, the S-Band steerable communications antenna must be stowed in a retracted position. During LEM checkout this antenna is extended by mechanical means to its operating position.

Varying 1 to 3 inches from the cabin pressure shell, and external to it, is a covering of thin gage aluminum. Insulation consisting of multiple layers of aluminized Mylar is installed between these surfaces. This forms the thermal shielding necessary to provide, in conjunction with the ECS, acceptable temperatures within the LEM cabin, unpressurized equipment bays and tank areas. This shielding also provides some meteroroid protection.

8.3.2 Descent Stage

The descent stage structure provides the supporting points for securing the LEM within the S-IVB adapter. It also supports the landing gear and provides a launch pad for the ascent stage at lunar liftoff. The landing gear, because of space limitations within the S-IVB adapter, is held by four electroexplosive devices (one for each leg) in a stowed configuration. During LEM checkout these devices are

energized thru the redundant pyrotechnic circuit and, after initiation, release the four legs. After release, the legs are spring-driven to their extended position and automatically locked.

Attenuation of landing loads is accomplished by the use of crushable metal cartridges confined within the primary and secondary struts. Pads at the foot end of each leg provide acceptable bearing on the anticipated lunar surface.

Structure is provided for support of the descent engine, fuel, oxidizer and helium tanks, landing radar antennae, scientific equipment, oxygen tanks, hydrogen tanks and water tanks. The engine is mounted in a separate compartment which is insulated to prevent excessive heat transfer from the engine during its operation.

The entire descent stage, except the surface exposed to engine heat, the landing radar, landing gear and exposed parts of the engine is enclosed by the same type of insulation used for the ascent stage.

No special provisions are made for meteoriod protection. Protection is that inherent with the existing structure and insulation.

8.3.3 Interstage Connection

Joining the ascent and descent stages together are four bolts and nuts which are units of a pyrotechnic system (see figure 8-2). Upon command, each bolt is broken and each nut is released (parts are contained) and the structural tie between stages is removed. The initiators in the bolts and nuts are energized by the current from two batteries and act simultaneously but either is capable of doing the

job separately. Also linking the two stages is an electrical and hard line umbilical which is led through two pyrotechnic devices which, when energized by the batteries, severs the bundle with a guillotine and linear shaped charge action. The hydrogen and oxygen stored in the descent stage flows through two quick disconnects in the area of the umbilical. These disconnects are spring-loaded mechanisms that self-seal when a depressing force is removed as happens at lunar ascent stage lift off.

SECTION 9

COMMUNICATIONS SUBSYSTEM

9.1 SUMMARY

The Communication Subsystem contingency analysis indicates that no single failure condition exists from which a safe abort cannot be achieved. The analysis reveals four single failure contingencies which may result in mission failure. The four contingencies are:

- failure of LEM VHF transceiver "A"
- failure of LEM VHF transmitter in transceiver "B"
- failure of the S-band steerable antenna
- failure of the premodulation processor (PMP) power supply

This contingency analysis depends to a great degree on the antenna gains, transmitter power and other design requirements contained in the LEM 10 Circuit Margin Analysis (LED 380-3). Since the circuit margin analysis is based on past requirements which have since changed, corrections to account for changes in the subsystem operating modes and hardware are required. These corrections are being made presently as part of a current circuit margin analysis. However, the results of this study are not yet available, with the result that portions of this contingency analysis may require revision at some future time. Important elements of the analysis are:

- Failure of the LEM VHF transceiver "A" results in interruption of direct voice communication between LEM and CSM. During phases in which the LEM is not in LOS with MSFN, the VHF transmitter "B" is normally required to transmit LEM status data to CSM and is therefore unavailable as a voice backup to transmitter "A". During all other phases however, the "B" transmitter is available as backup for voice use. During rendezvous operations, LEM LOS with MSFN does not exist with the result that no communication backup for lost VHF is presently available. The UHF "C" transmitter is idle

during this phase but cannot be used presently as a backup because its operating frequency is not compatible with the CSM receiving equipment. It is suggested therefore that a study be made to determine the feasibility for implementing a dual frequency capability within the "C" transmitter in order that a VHF backup be available during the rendezvous and docking phase.

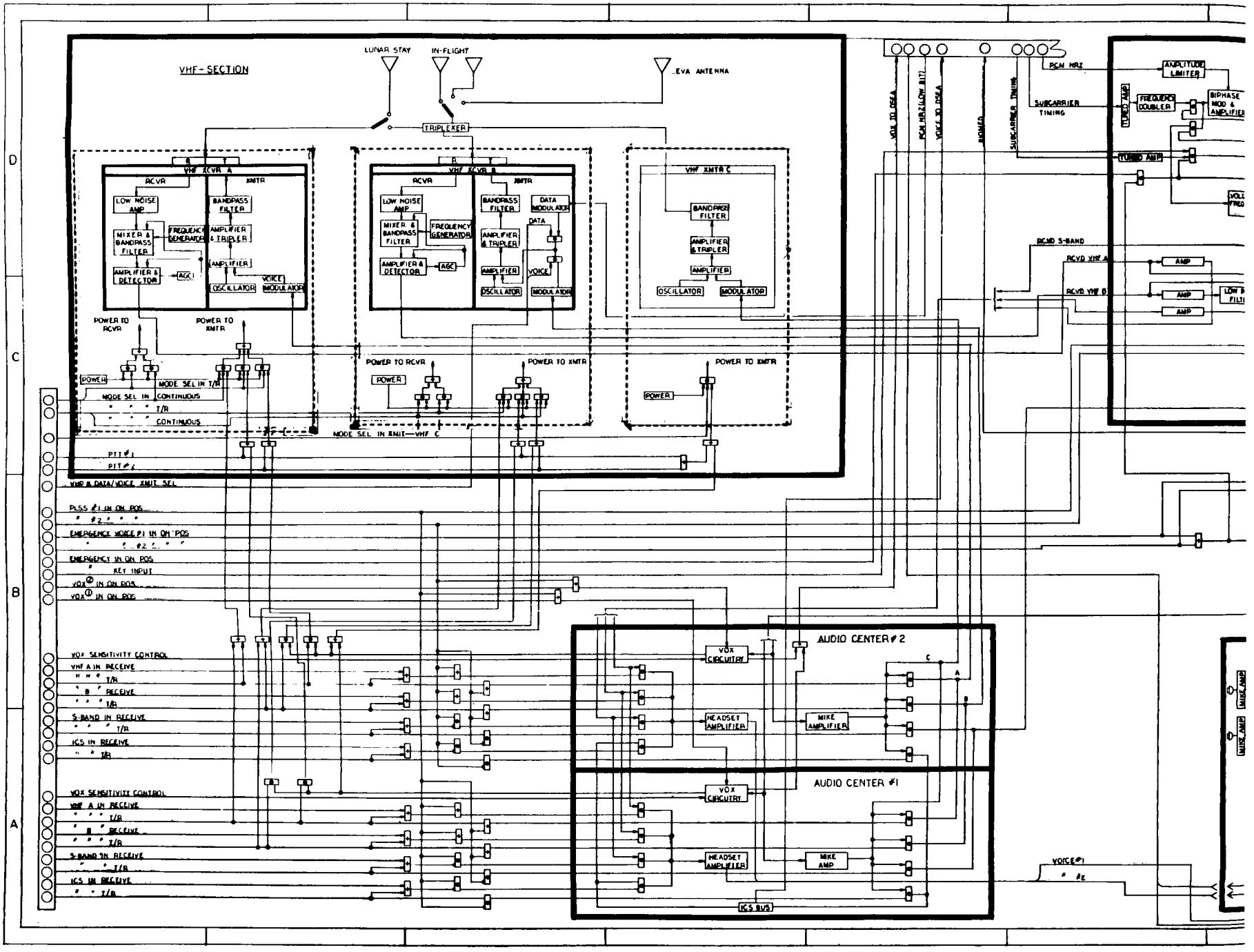
- Failure of the LEM VHF transmitter in transceiver "B" results in an interruption of LEM data which is normally sent to CSM for storage when LEM is not in LOS with MSFN. This data link is presently considered a mission success requirement. As such, its loss due to a single failure might necessitate an abort.

- Failure of the S-band steerable antenna to point properly to earth during in-flight phases results in loss of all LEM to MSFN functions except emergency voice and emergency key. The ability of the crew to manually perform antenna steering during the descent and ascent powered phases is highly questionable due to the high crew task loading associated with these phases. During coasting periods of the ascent or descent, it is possible that the antenna could be manually pointed. A minimum amount of manipulation by the crew would be required if the LEM were operated in a narrow band limit cycle mode (0.3 degree) during these phases because of the 6 degree beamwidth of the antenna. The present VHF data link (LEM to CSM) is no complete substitute for the S-band data link because of its low bit rate (1.6 kbs) and because it can be used only when a LOS to the CSM exists.

- Three separate sections within the PMP power supply can be failed singly. Failure of each section results in different functional losses. The least loss results from failure of regulator No. 1 wherein all S-band functions are retained with the exception of the low power FM lunar stay mode. The greatest loss occurs with a failure of regulator No. 2 resulting in loss of all S-band telemetry.

9.1.3 A

[redacted]



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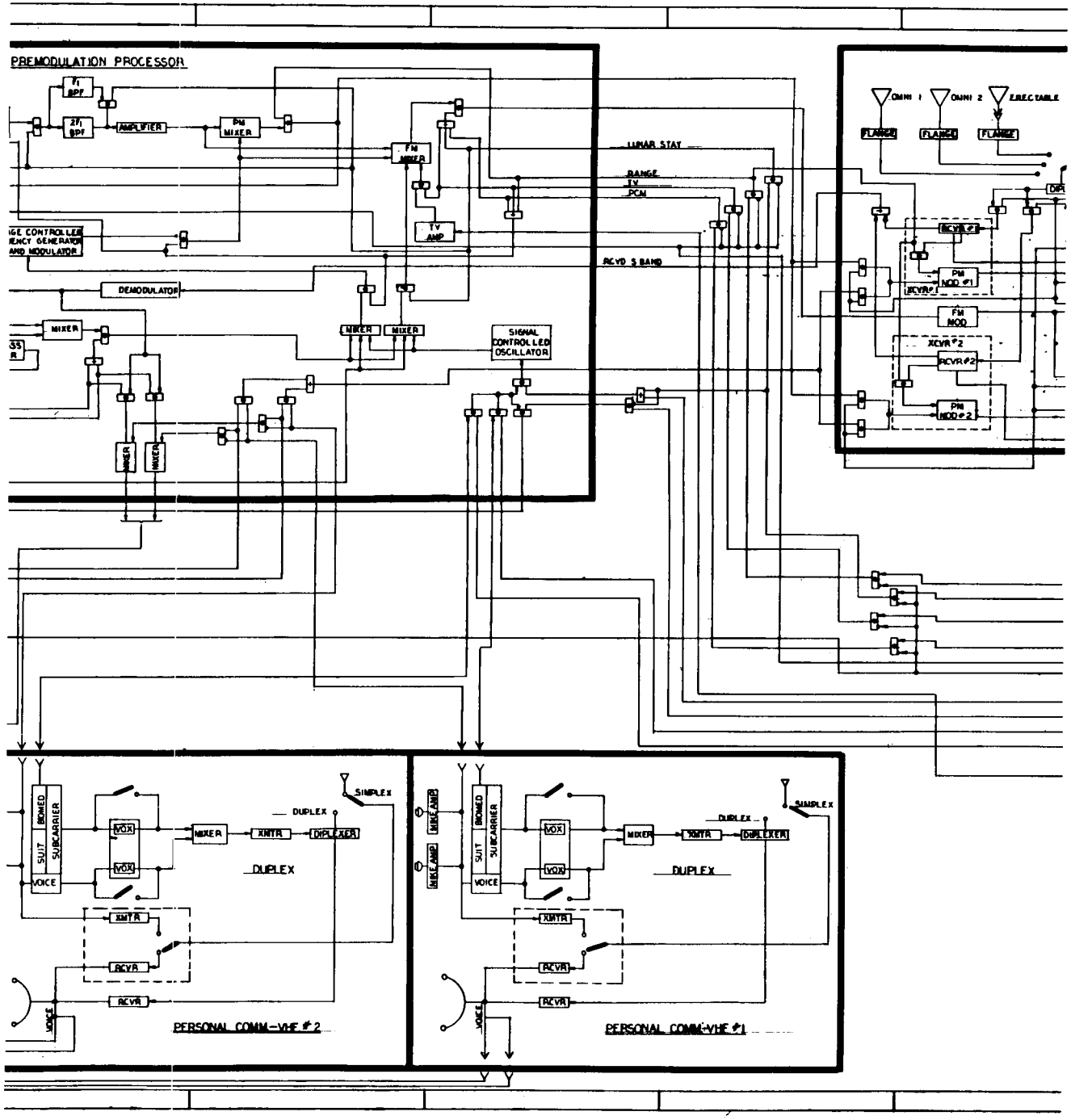
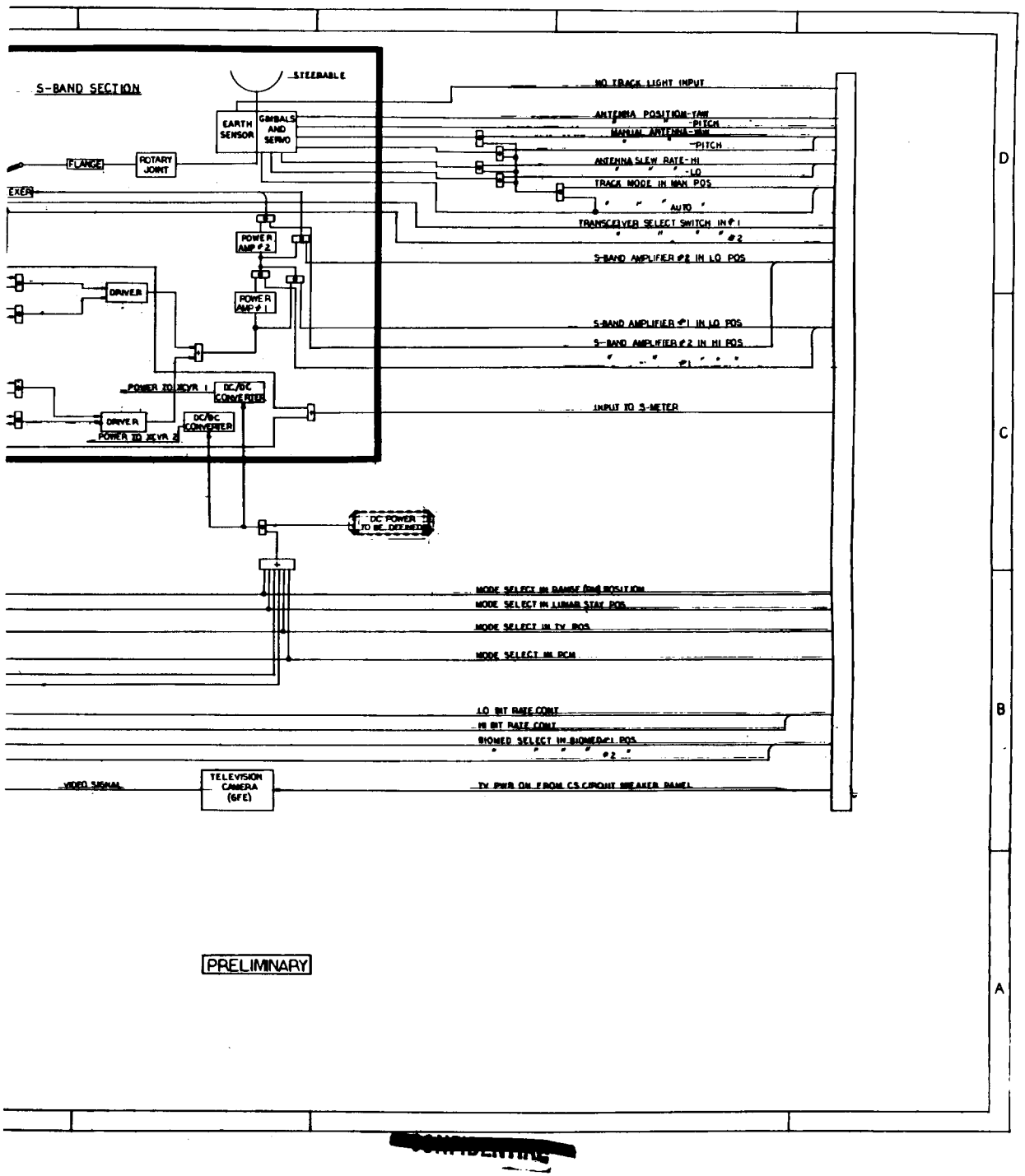


[redacted]

Figure 9-



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1 Functional Configuration, Communications Subsystem

SUBSYSTEM: Communications 9.2 WORKSHEETS *PHASE:* All LEM Phases

EQUIPMENT/ FUNCTION	*FAILURE MODE*	*UNCOMPENSATED FAILURE EFFECT*	*MISSION COMPLETION MODE*	*MISSION ABORT MODE*	*REMARKS*
Premodulation Processor (PMP) power supply	Failure of regulator #1 in the PMP power supply	Loss of lunar stay TV "off" mode only	Use lunar stay FM mode (TV, telemetry, voice, biomed) during lunar operations. During in-flight phases continue normally with PM mode.	Same as for mission completion.	
	Failure of regulator #2 in the PMP power supply	Loss of all telemetry and voice capability except emergency voice and key	Continue without LEM to MSFN telemetry link. (1) Voice link may be established by either S-band emergency voice or by VHF relay via CSM when in LOS.		VHF is unimpaired. (1) PCM LEM status data to MSFN is considered critical to mission success. This loss may cause mission to be aborted.
	Failure of regulator #3 in the PMP power supply	Loss of PM in-flight mode, FM TV lunar stay mode, FM backup (telemetry/voice) mode	In-flight: use lunar stay TV "off" FM mode with PA "on". Lunar stay: same except PA "off".		VHF is unimpaired. (1) Functions available are: • Voice • Low bit telemetry • Hard line analog biomed
Portable LEM TV, secondary emission conductivity vidicon (SEC tube)	Potassium chloride target within SEC tube disintegrates due to runaway condition	No TV from lunar surface or no backup for CM TV when used in CM cabin	Proceed without portable LEM TV camera.		Portable LEM TV low light level camera was primarily designed for on-lunar-surface TV presentations, hand-carried by EVA. MSC has subsequently designated LEM TV to be used additionally inflight within CM Cabin as backup for CM TV camera. Backup SEC tube substitution not practical. LEM TV camera has no mission success requirement status.
Microphone amplifier No. 1 in audio center transmit portion for Astro No. 1	Mike amp No. 1 circuit open; consequently, has no AF output	VHF transmitter voice path, ICS path or S-band transmitter voice path interrupted for Astro No. 1	Astro No. 1 unplugs his audio jack from Audio Center No. 1 and substitutes his second plug (which is normally not connected) thus automatically placing Astro No. 1 in parallel with Astro No. 2 through Audio Center No. 2.		Parallel operation requires that the two Astro's time share.
S-band Power Amplifier (PA#1) in S-band Transponder	Amplitron in PA#1 inoperative due to power supply PS#1 failure	Loss of all in-flight functions except emergency key and emergency voice; also loss of lunar stay functions as follows: • TV • hi-bit telemetry	Switch power supply #2 B ON for PA#2 redundant amplitron power amplifier.		The S-band Power Amplifier amplitron when missing B will act as a straight through path for the driving signal like a bandpass filter with a negligible insertion loss. Thus in tandem arrangement with PA#1 off and PA#2 on, normal power amplifier function is possible.

SUBSYSTEM: Communications

PHASE: All In-Flight Phases

EQUIPMENT/ FUNCTION	*FAILURE MODE*	*UNCOMPENSATED FAILURE EFFECT*	*MISSION COMPLETION MODE*	*MISSION ABORT MODE*	*REMARKS*
S-band, 2-ft steerable antenna (mounted on LEM)	Servos are inoperative in gimbal and servo section of steerable platform	Antenna repositioning to keep antenna pointing toward earth; as directed by the sensors, is now impossible as servos will not respond. This results in ineffectiveness of S-band steerable 2-ft antenna, incapacitating all S-band inflight LEM-to-MSFN communications. (1)	Switch to emergency backup inflight omni S-band antenna No. 1 or 2 mounted on LEM. Switch off all modes and use only emergency voice (link margin -1.0 db). Ranging function alone might be possible (no circuit margin available for this at present).	Same as for mission completion mode.	Emergency backup S-band inflight antenna, consists of two conical log spirals with a gain of -3 db. This means an approximate 23-db drop in gain from normal inflight S-band steerable antenna. For comparison, normal inflight mode (PM) link margins for ranging, telemetry and voice with the steerable antenna are: • PCM telemetry 5 db • Voice 9.4 db • Carrier 38.3 db • PRN ranging 24. db (1) All except emergency voice.
Remotely controlled range function switch located within the S-band transponder between the receiver and PM modulator	Ranging switch fails open; signal branch from receiver to PM transmitter is now interrupted	MSFN-LEM-MSFN tracking or ranging by the ground is not possible	CSM rendezvous radar has capacity to establish LEM's position relative to CSM. PRN ranging of CSM by MSFN is retained. Thus, LEM position relative to MSFN may be re-established.		Ranging signal gets to LEM S-band PM receiver but never gets beyond range switch to wideband input of PM transmitter where it is intended for rebroadcast to ground.
Portion of PMP	In PCM-NRZ data input branch from PCMTE to PMP: data AM limiter is shorted or the bi-phase modulator is inoperative	LEM-to-MSFN (PM mode) functions lost; serial PCM-NRZ digital data at hi-or-lo-bit rate does not pass thru.	Turn off PCM format switch. Restoration of hi-bit PCM LEM status data now becomes impossible. However, on lo-bit position of telemetry rate switch on control panel, direct analog hard-line biomed can be implemented. This restores biomed data of either Astro, one at a time (whoever is switched in), analoged on a 14.5-kc sub-sub-carrier and frequency modulated on the 1.25-mc subcarrier then applied to S-band PM input of S-band transceiver.		No amount of switching will restore missing hi-bit digital NRZ data. Biomed can be restored in analog form. Low bit rate relay via VHF to CSM is available during CSM-LEM LOS periods. However, PCM LEM status data to MSFN is considered critical to mission success. This may cause mission to be aborted.
	In 512-kc PCMTE subcarrier reference branch from PCMTE to PMP; 512-kc AM limiter in PMP is shorted	PM mode LEM-to-MSFN functions lost: • Hi-bit 51.2 kbs PCM telemetry including biomed and LEM status information, due to missing 1.024-mc subcarrier • Lo-bit rate, which is theoretically possible on 1.024-mc subcarrier, is also out	Restoration of the hi-bit 51.2 kbs PCM telemetry along with included biomed and LEM status information becomes impossible by switching to logical inflight FM backup mode, i.e., switching to Telemetry/Voice mode position, since 1.024-mc subcarrier is missing. To restore PCM telemetry, switch to a compromise lo-bit PCM telemetry in the narrow-band FM lunar stay mode position, voice/biomed, lo-bit telemetry, PA on. LEM status data is now on 512-kc subcarrier, and hard-line biomed electrocardiogram data on 14.5-kc subcarrier PRN ranging capability by MSFN is now unavailable, except when switching back to inflight mode, then losing PCM in turn.		The 512-kc subcarrier reference input to PMP from PCMTE when interrupted as depicted, causes a partial failure in all PMP-implemented operational modes except emergency voice and emergency key for this failure. All hi-bit PCM telemetry on all PMP-implemented mode positions is lost. Lo-bit PCM is available in lunar stay mode with TV off. Simultaneous voice and biomed also are available. PRN ranging is possible only by switching back to the defective inflight mode (when required to do so) but then PCM is lost. Ranging is meaningful only in PM mode.

SUBSYSTEM: Communications

PHASE: All In-Flight Phases

EQUIPMENT/ FUNCTION	*FAILURE MODE*	*UNCOMPENSATED FAILURE EFFECT*	*MISSION COMPLETION MODE*	*MISSION ABORT MODE*	*REMARKS*
PMP	In 512-kc branch from timing equipment to PMP; specifically, 512-kc tuned amplifier is open	LEM-to-MSFN PM mode functions lost: • Voice on 1.25-mc subcarrier due to loss of 1.25-mc VCO • Emergency keying due to loss of 512-kc subcarrier • Low bit telemetry on 512 kc sub-carrier	To restore voice capability, switch to inflight backup (FM mode emergency voice only). To restore hi-bit PCM telemetry, switch to telemetry voice position. To restore biomed, switch to FM lunar stay mode with PA "on".	Return to CSM with failure compensated as for mission completion mode.	The 512-kc (TE) input to the PMP, when interrupted as depicted here, causes a partial failure in all PMP-implemented operational modes except emergency voice. The nature of the failures are peculiar to the respective modes and only specific mission priority needs can dictate timely appropriate alternates. NOTE: The originally available hi-bit PCM/voice and tracking or ranging capability can no longer function simultaneously, only separately as follows: • PM mode hi-bit PCM and ranging only • FM mode (backup No. 1) emergency voice only • FM mode (backup No. 2) hi-bit PCM only • FM lunar stay mode, direct hardline biomed only

SUBSYSTEM: Communications *PHASE:* Lunar Stay

EQUIPMENT/ FUNCTION	*FAILURE MODE*	*UNCOMPENSATED FAILURE EFFECT*	*MISSION COMPLETION MODE*	*MISSION ABORT MODE*	*REMARKS*
Portion of PMP	In 512-kc branch from TE to PMP; the 512-kc tuned amplifier is open(1)	LEM-to-MSFN FM mode functions lost: • In narrowband FM PA off position (voice/biomed/lo-bit PCM), the lo-bit PCM (1.6 kbs) telemetry is lost. • In wideband FM position (TV/voice/biomed/hi-bit telemetry), the voice/biomed of both Astros lost while operating in lo-bit-rate mode. At hi-bit rate, voice/biomed of EVA is lost.	Restoring voice/biomed can be accomplished only at the expense of: • Loss of TV when switching from wideband FM to narrowband FM backup. • Using only PCM hi-bit or lo-bit telemetry and biomed of internal Astro only, with no voice. • Using emergency voice only, but not PCM and voice simultaneously.	Switch to PM inflight mode and return to CSM.	(1) Except for emergency voice, this failure leaves no other mode position completely intact. This forces singular use of partial modes on a time-shared basis. Loss of 512-kc subcarrier causes loss of emergency key. Changes in switching and implementation will be made in future to the PMP to accommodate MSC requested modifications to the LEM communications subsystem.
	In 512-kc PCMTE subcarrier reference branch from PCMTE to PMP; the 512-kc AM limiter in PMP is shorted	LEM-to-MSFN functions lost: 51.2 kbs hi-bit PCM telemetry with biomed from internal Astro and PCM LEM status data, or possible lo-bit PCM telemetry only, due to loss of 1.024-mc sub-carrier. This occurs in the TV (PA on) telemetry voice/biomed wideband FM lunar stay mode.	To restore PCM hi-bit telemetry becomes impossible. Must switch to narrowband FM voice/biomed, lo-bit telemetry (PA off) mode, and use available lo-bit 1600-bps rate on 512-kc subcarrier. Voices of EVA and internal Astro are available on 300-cps to 3-kc baseband. EVA biomed with spacesuit data are on seven channels from 4.0 thru 12.4-kc subcarriers. Hardline biomed electrocardiogram analog data from internal Astro now available on 14.5-kc subcarrier.		Mission completion can be affected only if compromise use of lo-bit rate telemetry in place of lost hi-bit-rate telemetry is acceptable.
	In PCM NRZ data input branch from PCMTE to PMP; data AM limiter is shorted	LEM-to-MSFN FM mode functions lost; serial PCM NRZ digital data at hi-or lo-bit rate do not pass through data AM limiter; hi-bit 51.2 kbs PCM telemetry on 1.024-mc subcarrier is missing, this includes biomed and LEM PCM status data	Turn off PCM format switch. Note restoration of hi-bit PCM 51.2 kbs telemetry LEM status data now impossible. However, in lo-bit position of telemetry rate switch on control panel, direct analog hardline biomed can be implemented. This restores hardline voice/biomed electro-cardiagram data of either Astro, one at a time (whoever is switched in), analoged on a 14.5-kc sub-subcarrier and FM'd on the 1.25-mc subcarrier, then applied to the S-band FM input of the S-band transceiver. The TV telemetry/voice-biomed-PA on Lunar Stay mode will be missing only LEM status data. Switching to voice/biomed lo-bit telemetry PA off mode will be normal except for the missing lo-bit PCM telemetry on 512-kc subcarrier.		Lunar stay mode will be adversely affected to the extent of losing LEM PCM data capacity. No switching can restore it. PCM LEM status data to MSFN are considered critical to mission success, this may cause the mission to be aborted. Low bit rate relay via VHF to CSM is available during CSM-LEM LOS periods.
PLSS transmitter operating in duplex mode	No RF output from PLSS transmitter	Loss of communications from EVA to LEM (i.e., 7-channel biomed and voice) thus also affecting conference monitoring by MSFN and CM	EVA return to LEM. Resume lunar exploration operations with second PLSS. (1)	EVA returns to LEM. Return to CSM.	(1) EVA could, with this failure, switch to simplex thus necessitating LEM VHF to operate with system "A". The present LEM lunar stay antenna radiation pattern is such that communication between EVA and LEM VHF "A" as backup to this contingency does not appear acceptable. Further study is required.

SUBSYSTEM: Communications

PHASE: Lunar Stay

EQUIPMENT/ FUNCTION	*FAILURE MODE*	*UNCOMPENSATED FAILURE EFFECT*	*MISSION COMPLETION MODE*	*MISSION ABORT MODE*	*REMARKS*
VHF transmitter "C"	No RF output from LEM VHF transmitter "C"	Loss of communication LEM to EVA	Preferred mode:(1) Astro inside LEM connects his PLSS antenna lead to LEM EVA antenna and continues in duplex mode. Alternate mode:(1) Switch VHF transmitter "A" to EVA antenna using existing triplexer and continue in simplex mode.	EVA returns to LEM. Return to CSM.	(1) In the event of this failure, EVA should have prearranged understanding to allow for some lapse of time while LEM Astro switches over. Present design does not include the suggested switching of PLSS into the LEM EVA antenna system. This technique requires design study. The preferred mission completion mode offered in preference to operating in simplex mode for the following reasons: • Operation in simplex does not provide EVA biomedical data • Operation in simplex with VHF transceiver "A" requires time sharing between EVA and CSM.
VHF transceiver "A"	No RF output from transceiver transmitter or inoperative CSM receiver	Loss of voice communication via VHF from LEM to CSM; MSFN monitoring via S-band unaffected	Use redundant VHF transmitter "B" or use earth as relay on S-band to CSM.	Same as for mission completion mode.	Present use of VHF "B" transmitter is primarily for LEM data transmission. Its use as backup to the VHF "A" requires further investigation.
EVA VHF PLSS duplex transmitter	No RF output from PLSS duplex transmitter or inoperative LEM VHF receiver "B"	Loss of voice communication from EVA to CSM via LEM	EVA must switch to PLSS simplex VHF operation and inform Astro inside LEM to switch transceiver "A" to EVA antenna and time share transceiver "A" between EVA and CSM.	EVA returns to LEM. Return to CSM.	EVA cannot be heard by Astro inside LEM after switching to simplex since transmitter frequency is not the same as before. LEM Astro transmitter frequency can no longer be heard by EVA on simplex receiver, thus necessitating LEM Astro to use his simplex gear.
VHF receiver "B"	Fails to detect EVA transmitted signal	Loss of communication EVA to LEM	Continue without EVA to LEM link or Astro inside LEM connects his PLSS antenna lead to LEM inflight antenna; PLSS receiver substitutes for failed LEM receiver.		In the event of this failure, EVA should have prearranged understanding to allow for some lapse of time while LEM Astro switches over. Present design does not include the suggested switching of PLSS into the LEM inflight antenna. This technique requires design study. See also remarks for transmitter "C" failure.

SUBSYSTEM: Communications

PHASE: Lunar Orbit Prior to Separation and Final Dock Thru TEI

EQUIPMENT/ FUNCTION	*FAILURE MODE*	*UNCOMPENSATED FAILURE EFFECT*	*MISSION COMPLETION MODE*	*MISSION ABORT MODE*	*REMARKS*
LEM S-band transceiver PM transmitter	No RF output from PM transmitter/driver #1 (multiplier/buffer amplifier chain #1)	Lose LEM-to-MSFN inflight S-band voice-biomed communication; 51.2 Kbs hi-bit PCM telemetry; pseudo random noise (PRN) tracking and ranging	Switch to redundant multiplier/buffer amplifier driver chain #2 to restore all functions.	Jettison LEM and return to primary earth landing site.	In PRN ranging operation, PM mode is needed to maintain phase coherence. This is necessary to guarantee "accuracy" in ranging. In another mode, ranging results are meaningless to MSFN.
LEM VHF transceiver transmitter "B"	No RF output from LEM VHF transceiver Data transmitter"B"	Lose LEM to CSM inflight PCM real time data transfer capability intended for CSM recorder when LEM is not in LOS with MSFN	Continue mission without PCM data transfer to CSM.	Same as above	Presently the LEM to CSM VHF data link is considered a mission success item. As such, this failure would probably necessitate an abort.

SUBSYSTEM: Communications

PHASE: Rendezvous and Docking

EQUIPMENT/ FUNCTION	*FAILURE MODE*	*UNCOMPENSATED FAILURE EFFECT*	*MISSION COMPLETION MODE*	*MISSION ABORT MODE*	*REMARKS*
VHF transceiver "A" inoperative	No d-c power to VHF transceiver "A" due to short circuit	Lose VHF communications between LEM and CSM(1)	Continue without VHF voice capability(1)	Same as mission completion mode.	(1) Idle transmitter "C" cannot be used as backup because its frequency is not compatible with the CSM receive frequency. Transmitter "B" is unavailable because it is normally used in this phase for data transmission. It could be time shared however between data and voice if interruption of the LEM status data were acceptable.

Communications Subsystem (Cont.)

9.3 DESCRIPTION

The Communications Subsystem provides the link between the astronauts, the orbiting Command and Service Modules (CSM), and earth monitoring stations. The subsystem contains two radio frequency (RF) sections, one operating in the VHF range and the other in the UHF range (S-Band), a television section, and a signal processing section (see Figure 9-1).

The RF sections provide two-way voice communications between the LEM and the CSM, LEM and earth, LEM and the extravehicular astronaut, and between crewmembers within the LEM. In addition, these sections receive and transmit tracking and ranging information from earth and transmit to earth television telemetry, and bio-medical information. Emergency voice is provided as a backup for failed normal voice. Emergency keying, a second back up, is provided for failed emergency voice.

The television section is utilized by the extravehicular astronaut to televise pictures of the lunar surface. An eighty-foot TV cable is attached to the landed vehicle.

The signal processing section modulates, switches, amplifies, and multiplexes the various signals and inputs from other subsystems during the different operating modes.

9.3.1 S-Band Section

The S-band section provides the LEM-earth communication link. It comprises an S-band transponder assembly, two power amplifiers, a diplexer, an RF switch, a steerable antenna with appropriate control

and servo circuits, an erectable antenna, and an emergency omni-directional antenna. Television, voice, telemetry, biomedical, emergency voice, emergency keying, and PRN tracking and ranging signals are accommodated by this equipment.

9.3.1.1 S-Band Communications: The S-band communications section receives information from the PMP, in some cases directly, and in other cases in response to control settings made by the astronaut. This information is applied to the inputs of the S-band transponder assembly, a solid-state device consisting of the following:

- two phase locked receivers
- two transmitter chains to provide redundancy
- two power supplies
- one FM/PM capability with switching and isolation
- two PM modulators

The information is transmitted from one of the transmitter assemblies of the transponder to either of two power amplifiers connected in tandem. The information is then conveyed to the S-band diplexer, which permits simultaneous reception and transmission from a single antenna, and then through the RF switch to one of the three S-band antennas. When less gain is required, the power amplifier in use is effectively bypassed and the driver signals routed directly to the antenna.

Signal reception from earth is confined to voice and pseudo-random noise (PRN) ranging signals. The signal flow is from the antenna through the RF switch, diplexer, phase-locked receiver, premodulation processor, audio center, and to the astronaut's headset or through the VHF transmitter to the roving astronaut on the lunar surface. All S-band communications are accomplished during line-of-sight phases of the mission.

9.3.1.2 Steerable Antenna: The steerable antenna is a medium gain, 2 ft. parabolic reflector. It is mounted to the LEM structure and is used during the LEM lunar orbit, descent, ascent, rendezous, and docking phases of the mission. The antenna is fastened on a double elevation gimbal that is servo-controlled in response to automatic or manual slewing signals. Switches on the control panel provide mode selection and slewing commands to the antenna servos. An infrared earth sensor provides tracking error signals in the automatic mode.

9.3.1.3 Erectable Antenna: The erectable antenna is a high-gain unit comprising a 10 ft. parabolic reflector mounted on a tripod support with an aiming telescope.

This antenna is used during the lunar stay to transmit and receive S-band communications between LEM and earth.

9.3.2 VHF Section

The VHF equipment provides the LEM - EVA and LEM - CSM communication links. Included are the VHF/AM transceivers A and B, VHF transmitter C, a VHF triplexer, an RF switch, an in-flight antenna, a lunar-stay (bifilar helix) antenna, and an EVA antenna.

9.3.2.1 VHF Communications: The VHF communications link provides two-way voice communications during line-of-sight phases of the mission between the LEM and the CSM and the LEM and the extravehicular astronaut. It is also used for data transmission from LEM to the CSM while in flight behind the moon. Voice transmissions are processed through the audio centers and are coupled to the VHF/AM transceivers, i.e. VHF transceiver A is primarily used for voice transmission between LEM and CSM in the simplex mode. VHF transceiver B is used for sending LEM data while behind the moon to the CSM. Receiver B is used to receive duplex normal mode voice from EVA's PLSS transmitter. VHF transceiver A can be used as a simplex back-up for LEM-EVA voice communications (the lunar stay antenna pattern however may pose serious restraints in this use). VHF transmitter C is used in conjunction with the EVA antenna, for voice transmission from LEM to the EVA. Transceivers A and B share the VHF inflight antenna while space - borne. Transceiver A uses the lunar stay antenna and receiver B uses the inflight antennas while on the lunar surface. From the transceivers A and B, the signals are coupled to their respective in-flight or lunar-stay antennas through the VHF multiplexer and RF switch. Voice transmissions from the CSM are coupled from the appropriate antenna to the receiver portion of the transceiver A through the RF switch and the multiplexer. The receiver outputs are processed in the PMP and the audio centers and fed to the astronaut's headsets or routed back to the audio centers for VHF transmissions to the extravehicular astronaut. All LEM-CSM communication is accomplished during LEM-CSM line of sight phases of the mission.

Communications between the LEM and the EVA are accomplished through the VHF section. The VHF transceiver B receives voice and biomedical information concerning the physiological functions of the

astronaut and the operation of his PLSS while he is conducting lunar exploration. Voice information is monitored by the astronaut in the LEM and voice and biomedical information is re-transmitted via the S-band link to an earth monitoring station.

9.3.3 Signal Processing Section

The signal processing section consists of the premodulation processor (PMP) and two audio centers. The section receives signals from the Instrumentation Subsystem, the VHF and S-band sections, and processes them for transmission.

Specifically, the signal processing equipment provides the interface components for processing signals to be transferred between the various data gathering equipment external to and within the LEM, and the various receivers and transmitters. The processing functions include multiplexing, filtering, amplifying, and switching of voice, and television signals.

9.3.3.1 Premodulation Processor: The PMP contains filters, subcarrier oscillators, matching networks, the voltage controlled oscillator and bi-phase modulator, and switches. It performs signal modulation and mixing and accomplishes signal switching so that the correct intelligence corresponding to a given mode of operation is transmitted.

9.3.3.2 Audio Centers: The audio centers are two identical assemblies. These units operate with complementary panel controls to provide selection, isolation, and amplification of LEM audio to and from the subsystem transceivers.

9.3.3.2.1 Intercommunications: A closed circuit intercommunictions system, part of the audio centers, permits voice communications between astronauts when both astronauts are aboard the LEM.

9.3.4 Personal Communication System

The personal communication system is part of the astronaut's space suit. The duplex mode is used for transmitting and receiving voice. The emergency mode operation is simplex, voice only. Biomedical and suit data are sent along with the voice from EVA to LEM in the normal mode.

The system contains two transmitters and two receivers, one set for each mode of operation, and components such as mixer, amplifier, subcarrier oscillators, and diplexer, which are associated with the duplex and simplex modes. The output of the system is transmitted via the suit monopole antenna.

9.3.5 Television Section

The present LEM TV Section provides a capability for processing, and transmitting television pictures to earth while the LEM is on the lunar surface. Moreover the portable LEM TV camera is also used inside the command module, being transferred to the LEM prior to descent and returned to the CSM after ascent for use in the return phases of the mission.

Inside the command module cabin, the LEM portable TV camera is connected by means of a 5 ft. cable plugged into any of 3 positions provided for it. In lunar orbit the LEM TV will be transferred to the LEM

prior to LEM descent where it will be stored in the astronaut's cabin. After lunar landing, it will be handcarried on the lunar surface by the astronaut, or clamped at times to a LEM leg aimed at his activities, at the end of the 80 ft. TV cable plugged into the LEM. The LEM TV uses a water boiloff cooling system and is capable of operating during lunar day or night (plus or minus 4 days from maximum earth shine).

SECTION 10

INSTRUMENTATION SUBSYSTEM

10.1 SUMMARY

In the contingency analysis of the Instrumentation Subsystem, two aspects have been considered:

1) the performance of the several assemblies of the Instrumentation Subsystem dependent on the operation of other LEM susbsystems

2) the performance of the LEM subsystems dependent on the operation of the Instrumentation Subsystem

The following observations pertinent to this analysis have been made:

- Instrumentation serves as a medium for verification of proper subsystems operation. Malfunction of a measurement device could lead to erroneous information regarding spacecraft status. Thus, the Instrumentation Subsystem must be designed to generate as little doubt as possible as to the validity of its operation.

- Some of the outputs from Instrumentation are furnished to more than one destination, such as cockpit display, caution and warning devices, and PCM telemetry. System malfunctions indicated by such an output could conceivably be checked by comparing the flight display with the telemetered data using voice communication. Thus, faults in Instrumentation occurring after signal conditioning may be detected. Such a procedure would necessarily be based on the importance of the monitored parameter and on the phase of the mission.

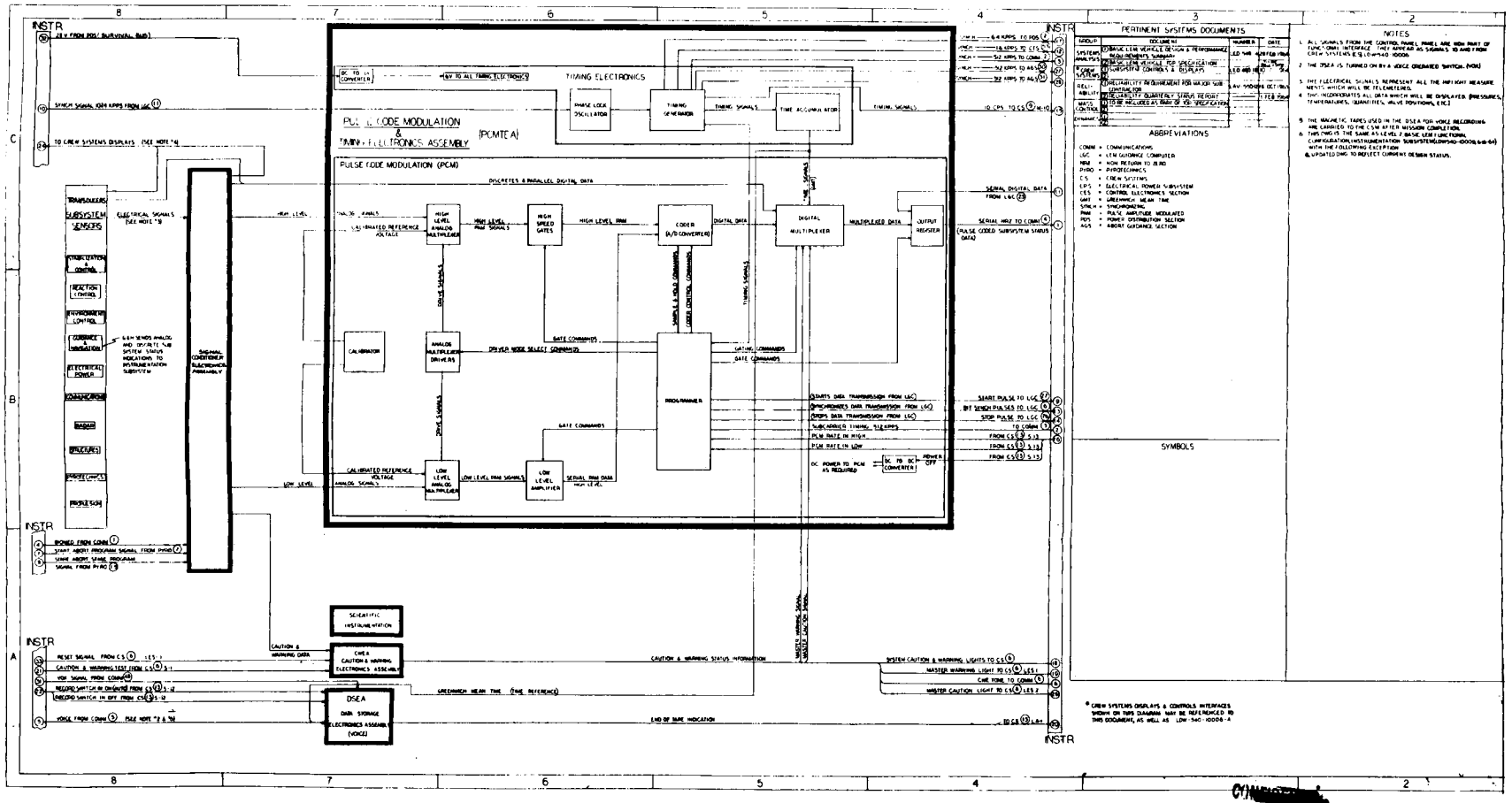
- Some of the outputs from Instrumentation are furnished to a single destination; for example, cockpit display or a caution and warning device. Depending on the parameter, a true out-of-tolerance condition might be accompanied by out-of-tolerance indications in other measurements, thus providing some means of verification. In general, malfunctions indicated by the Instrumentation sensors should be considered valid, despite the possibility that the malfunction might be in instrumentation.

- No single failure condition can occur within the Instrumentation Subsystem from which a safe abort cannot be achieved.

10.1.2.A

[redacted]

10.1.2.B



[redacted]

Figure 10-1 Functional Configuration, Instrumentation Subsystem

Instrumentation Subsystem (Cont.)

10.2 DESCRIPTION AND ANALYSIS

The Instrumentation Subsystem consists of the following equipments:

- Transducers
- Signal Conditioning Electronics Assembly (SCEA)
- Pulse Code Modulation and Timing Electronics Assembly (PCMTEA)
- Data Storage Electronics Assembly (DSEA)
- Caution and Warning Electronics Assembly (C&WEA)

10.2.1 Transducers

Transducers are located throughout the LEM to sense physical phenomena and to generate representative electrical indications for use by the Instrumentation Subsystem and by Displays. The transducers include but are not limited to temperature and pressure transducers, position transducers, velocity and acceleration transducers, vibration and acoustical transducers, flow and quantity gauging transducers, current transducers and switches.

In general, failure of an individual device results only in the loss of the individual measurement without causing additional failures. Exceptions to this are the following:

- Differential pressure transducers - rupture of the diaphragm causes contamination of the fluid under measurement by the internal fluid of the transducer; this internal fluid (e.g.

flurolube) is inert with respect to the measurand. However, the diaphragm is designed to withstand more than 5 times its maximum rated pressure; failure would occur well beyond the limit of allowable system operation.

- Snap-action switches - internal failure of a switch in "open" or "closed" condition will generate a constant "off" or "on" indication respectively.

10.2.2 Signal Conditioning Electronics Assembly (SCEA)

The SCEA performs the following functions:

- Acts as a junction box for all signals which aid in preflight assessment of spacecraft readiness, which aid in in-flight management of the spacecraft or which are telemetered for in-flight and post-flight evaluation of performance.
- Supplies precision power and routes raw power to the transducers.
- Conditions to proper form, level and impedance all sensor outputs not complying with the general LEM data acquisition requirements.
- Provides buffering and isolation, when required, for all signals.
- Routes all signals to proper destinations.
- Provides GSE interface connections for those signals not available at the PCMTEA output or at other GSE interfaces.

The SCEA does not supply or route power to transducers not part of the Instrumentation Subsystem, i. e., those required for automatic internal subsystem control.

A failure within a subassembly of the SCEA will result in the loss of valid data from an individual measurement source, with the following exceptions:

- Phase demodulator - a short-circuit of the input reference voltage in one demodulator will result in loss of all demodulator channels.
- Precision power supply input open circuit failure will result in loss of up to 10 channels.

Fail-safe buffering is incorporated whenever a single SCEA channel supplies several loads. A short circuit of one load will not result in loss of that channel to the other load(s).

10.2.3 Pulse Code Modulation and Timing Electronics Assembly (PCMTEA)

The PCMTEA combines low-level and high-level analog data, parallel-digital data and serial-digital data into serial, binary-coded, digital signals for the pre-modulation processor equipment (part of Communications Subsystem) and pre-launch checkout equipment (ACE). The PCMTEA also provides timing frequencies to synchronize other LEM equipment and binary-coded-decimal Greenwich Meridian Time for telemetering and for time-correlation of recorded voice data.

10.2.3.1 Timing Equipment (TE) portion of PCMTEA: The TE portion of the PCMTEA contains a timing equipment oscillator, a timing generator and output buffers, a time accumulator and power supplies. The timing equipment oscillator provides a back-up signal in case of failure of the timing signal input from the Primary Guidance and Navigation Subsystem (PGNS). During normal functioning of PGNS, the TE oscillator is phase-locked to the input timing signal. If the PGNS signal fails, the TE oscillator continues to furnish the TE with a timing reference signal whose stability is 2 parts in 10^6. When the PGNS signal is present, failure of the TE oscillator has no effect. When the PGNS signal is absent, failure of the TE oscillator results in failure of the entire PCMTEA including all TE outputs.

The timing generator and output buffers furnish timing signals to the PCM portion of the PCMTEA and to other LEM subsystems. The input or the internal reference timing signal is counted down to

generate the various frequencies needed. Failure of any timing generator stage also results in the failure of all lower frequencies including the time accumulator. Failure of any output buffer does not affect any other TE output. Outputs provided by the timing generator are:

- 512 KC (2 lines) to Abort Guidance System (AGS) of the Stabilization and Control Subsystem. Loss of these signals during normal operation of PGNS has no effect. In the event of PGNS failure, these signals furnish primary timing for the AGS. Failure of one line causes failure of the Abort Electronics Assembly; failure of the other line causes failure of the Abort Sensor Assembly.
- 512 KC to emergency keying function of the pre-modulation processor (part of Communications Subsystem). Failure of this signal results in loss of hand keying capability used in emergency communication operation and in loss of the 1.25 mc subcarrier. Low-bit-rate data could be transmitted on the 1.024 mc subcarrier instead of the 512 kc subcarrier.
- 6400 cps to power inverters of Power Generation Section of Electrical Power Subsystem. This signal holds the AC power frequency to 400 cps ± .0008 cps. Failure of this signal results in degradation of the power frequence to 400 cps ± 10 cps, causing degradation of the data storage equipment.
- 1600 cps to the power supply of the Stabilization and Control Subsystem. Loss of this signal during normal operation of PGNS has no effect. In the event of PGNS failure, loss of this signal results in loss of synchronization to 800 cps ± .0016 cps. The S&C Subsystem continues to operate with an internal frequency range of 720 to 800 cps.
- 10 cps to the event timer of the Displays and Controls Subsystem. If this signal is used, it will provide primary drive to the time-to-event display. Loss of this signal would cause failure of this display.

The PCMTEA time accumulator generates a parallel, binary-coded-decimal time code representing day of month and time of day. Failure of the time accumulator results in loss of the real time reference in the telemetered data; this is not a significant failure in that a real time base can be furnished by the ground computer. Failure of the time accumulator also results in loss of time correlation information on the voice recorder; this does not affect mission success or crew safety.

The TE has two redundant power supplies independent of the power supply for the PCM portion of the PCMTEA. Failure of either TE power supply or failure of the PCM supply does not affect any of the TE outputs. Both TE supplies are fed by the EPS survival bus.

10.2.3.2 PCM Portion of PCMTEA: The PCM accepts low-level and high-level analog data and parallel-digital-event data from the SCEA, serial-digital data from PGNS and parallel digital data from other LEM equipment. This information is multiplexed to form the PCM data format.

In general, failure of any PCM functional block will cause loss of multiple channels of information in the telemetered data. Identification of the failed functional block can, with the following exceptions, be derived from the configuration of the failed channels.

- An "open" input gate in the high-level or low-level analog multiplexer will cause loss of a single channel. Such a failure could possibly be verified from the earth by voice communication with the LEM.
- An "open" input gate in the digital multiplexer will result in loss of one bit in a parallel-digital data channel or of one channel of parallel-digital-event data.
- A "shorted" input gate in the digital multiplexer will result in loss of an 8-bit parallel-digital-data channel. Since a parallel-digital-event word contains 8 independent channels, failure of the entire word can be attributed to the PCM.
- Failure of the serial digital input gate or the serial data start-bit sync-stop pulses will result in loss of all serial digital data from PGNS.

10 2.4 Data Storage Electronics Assembly (DSEA)

A voice recorder failure results in the loss of recorded data for later reference but does not affect either crew systems displays or real-time telemetered data.

External factors affecting operation of the DSEA include a real-time reference from the time accumulator of the PCMTEA and the frequency stability of the AC power from EPS/PGS. A failure of the time accumulator results in loss of the time information for time-correlated voice recording. In the event that the frequency of the EPS/PGS changes from 400 cps at 2 x 10^6 primary stability to 400 at ± 10 cps secondary stability, the DSEA will operate in a degraded mode.

10.2.5 Caution and Warning Electronics Assembly (C&WEA)

The C&WEA provides the crewmen with a rapid check of the LEM status by continually monitoring critical subsystem parameters during the manned phases of the LEM mission. The C&WEA indicates the following:

- Caution functions:
  - Functions not requiring immediate attention but which would affect crew safety if action is not taken.
  - Malfunctions in subsystems which time-share displays.
  - Malfunctions in subsystems whose displays are not readily visible to the astronaut.
- Warning function - malfunctions which affect crew safety and require immediate attention.

All inputs to the C&WEA, except for electrical power and mode identification signals are furnished by the SCEA. Loss of an input signal due to SCEA or C&WEA malfunction results in loss of that one channel only, with two exceptions:

- Short-circuit failure of an input channel within the C&WEA may result in loss of up to four additional channels.

- Loss of an inhibit signal may cause erroneous malfunction indications. The inhibit signals are used to prevent misleading malfunction indications. These may arise in cases where a malfunction condition for one mode of operation of a subsystem is normal operation for another mode. Determination of inhibit signal failure can be obtained from examination of the actual mode of operation.

The C&WEA is designed for fail-safe operation in that an internal failure does not generate a malfunction indication.

SECTION 11

CREW PROVISIONS

11.1 SUMMARY

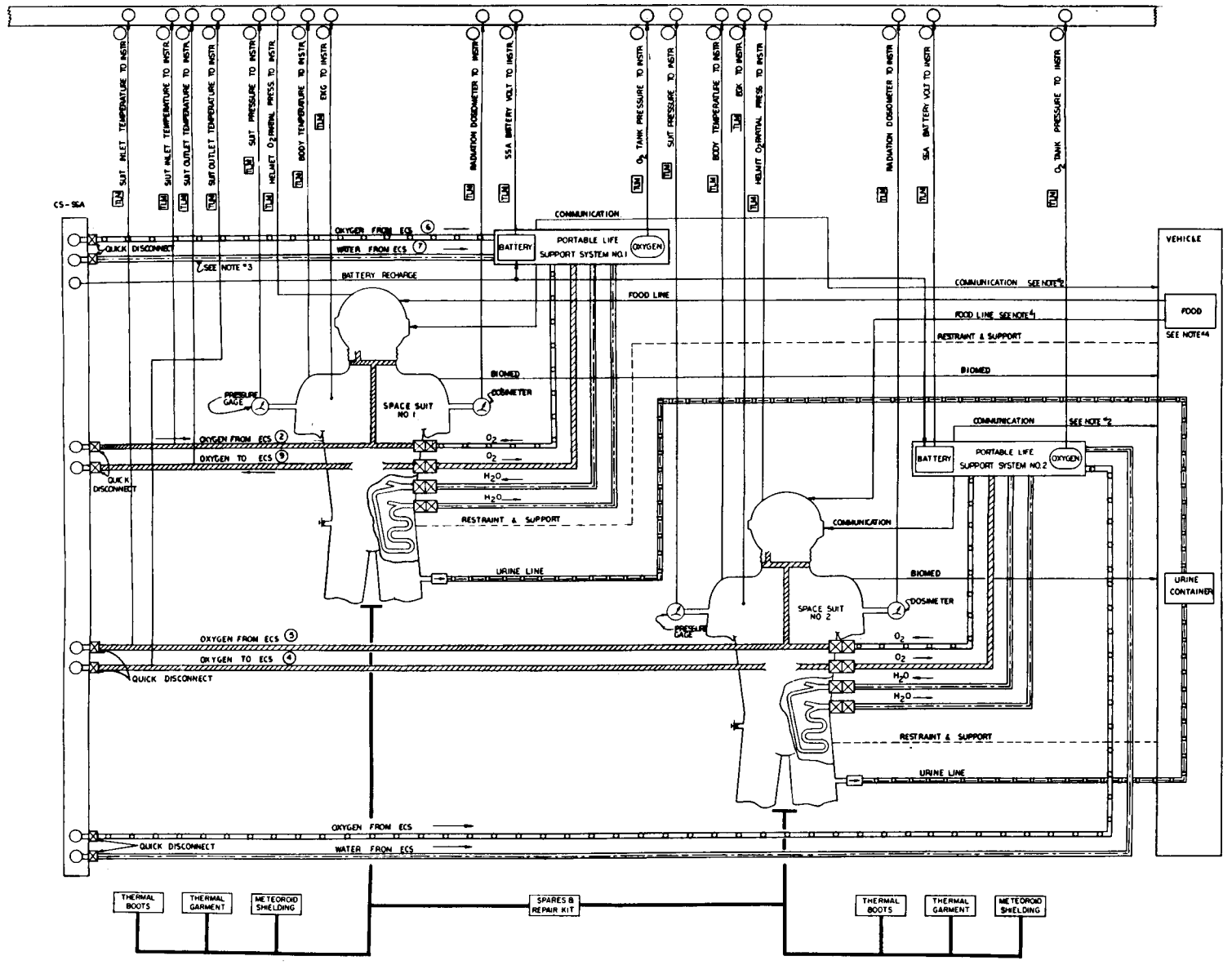
The LEM internal lighting contingency analysis included instrument and cabin illumination while the external lighting contingency analysis covered LEM docking light provisions. Both analyses indicate that in all LEM mission phases there is no single lighting failure for which there is no back up mode, and no failure which does not permit a continued mission.

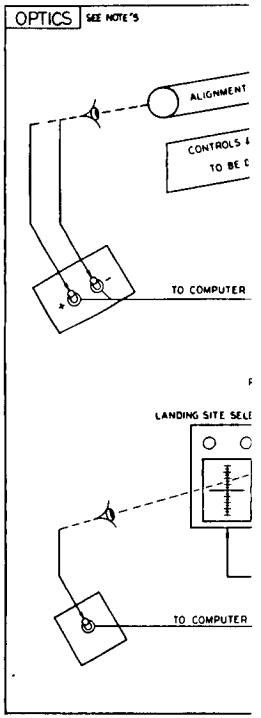
The contingency analysis for the umbilical hose interface between the PGA and LEM ECS indicates that in all LEM mission phases following separation, no umbilical hose failure will cause loss of life, or mission abort. The analysis further shows that at present the valve in the 72" umbilical hose provides no useful operational function if the use of the hose for lunar surface "buddy" rescue is not further implemented.

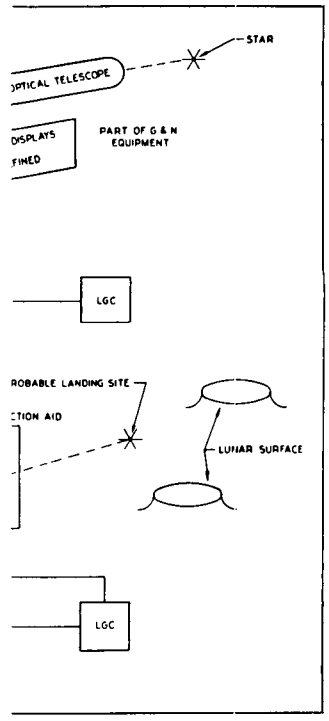
The failure of the forward hatch lock mechanism closed (see Structural/Mechanical worksheets, Lunar Stay phase) would prevent crew egress or ingress for lunar exploration. It suggests the need for other means to descend from LEM's upper surfaces if the upper hatch is to be used for egress instead of the forward hatch.

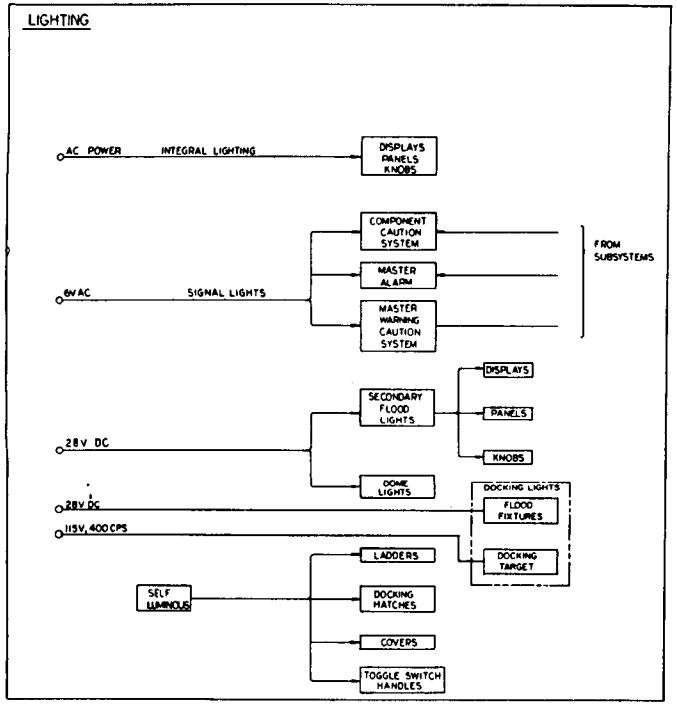
[redacted]

11.1.2-A









	RATED CAPACITY OF BRIGHTNESS	BRIGHTNESS ADJUSTMENT
FT. LAMBERTS UNLESS OTHERWISE SPECIFIED		TO BE SPECIFIED
1. DISPLAYS & PANELS PRIMARY INTEGRAL WHITE LIGHTING	0.5 ± 0.2	CONTINOUS FROM 0 FT. LAMBERTS
2. DISPLAYS & PANELS SECONDARY FLOOD WHITE LIGHTING	0.7 ± 0.3	" " " CANDLES
3. CHART READING & GENERAL ORIENTATION WHITE DOME FLOOD LIGHT.	( )*	" " " LAMBERTS
4. ILLUMINATED KNOBS & PUSH BUTTONS INTEGRAL WHITE LIGHTING	0.5 ± 0.2	" " " CANDLES
5. CAUTION SYSTEM (YELLOW) ANNUNCIATORS	50 ± 10	" " 15 ± 0.5
6. WARNING SYSTEM (RED) ANNUNCIATORS	50 ± 10	" " 15 ± 0.5
7. SUPPLEMENTAL MODULE LUMINATION SELF ILLUMINOUS	0.05 ± 0.02	NONE
8. MASTER ALARM (RED) LIGHTS	150 ± 30	NONE
9. COMPONENT CAUTION (YELLOW) LIGHTS	15 ± 3	CONTINUOUS FROM 0.45 ± 0.15
10. ILLUMINATED DISPLAYS (EL NUMERICS ETC.)	15 ± 3	" " "
	* FT CANDLE ON PANEL SURFACE	

Figure 11-1 Functiona

[redacted]

PERTINENT SYSTEM DOCUMENTS

GROUP	DOCUMENT	NUMBER	DATE
SYSTEMS ANALYSIS	(1) BASIC LEM VEHICLE DESIGN PERFORMANCE REQUIRMENT SUMMARY	LED-540-4	28 FEB 1964
	(2) BASIC LEM VEHICLE TOP SPECIFICATION		
CREW SYSTEMS	(1) SUBSYSTEMS CONTROLS & DISPLAYS	LED 480-14	10 JAN 1964
	(2) HUMAN ENG REQM'TS FOR LEM ILLUMINATION	LED 480-7	25 SEPT 1963
RELIABILITY	(1) RELIABILITY REQMT FOR MAJOR SUB CONTRACTOR	LAV 550 12	18 OCT 1963
	(2) RELIABITY QUARTERLY STATUS REPORT		1 FEB 1964
MASS CONTROL	1 TO BE INCLUDED AS PART OF TOP SPECIFICATION ABOVE.		
DYNAMICS			

NOTES

1. THE CREW RESTRAINT SYSTEMS HAS NOT BEEN COMPLETELY DEFINED AT THIS TIME HOWEVER AN UPRIGHT RESTRAINT SYSTEM RATHER THAN SEATS WILL BE ADOPTED.
2. THIS INTERFACE IS SHOWN IN LDW 540 10005 COMMUNICATIONS.
3. ONLY ONE PLSS AT A TIME CAN BE REPLANISHED WITH OXYGEN WATER OR BATTERY ENERGY.
4. FOOD IS MIXED WITH WATER BEFORE EATEN.
5. THIS IS MIT/IL EQUIPMENT.

ABBREVIATIONS

- O_2 - OXYGEN
- ECS - ENVIRONMENTAL CONTROL SUBSYSTEM
- CS - CREW SYSTEMS
- AC - ALTERNATING CURRENT
- DC - DIRECT CURRENT
- LGC - LEM GUIDANCE COMPUTER
- PDS - POWER DISTRIBUTION SECTION
- EL - ELECTRO LUMINESCENCE
- SSA - SPACE SUIT ASSEMBLY
- H_2O - WATER
- G&N - GUIDANCE & NAVIGATION
- TLM - TELEMETERED
- INSTR - INSTRUMENTATION
- EKG - ELECTROCARDIOGRAM

SYMBOLS

- CHECK VALVE
- PRESSURE RELIEF VALVE
- DISCONNECT
- OXYGEN FLOW LINE
- CONDITION OXYGEN FLOW LINE
- URINE FLOW LINE
- WATER FLOW LINE

FUNCTIONAL CONFIGURATION
CREW SYSTEMS

Figure 11-1

d Configuration, Crew Systems [redacted]

SUBSYSTEM: Crew Provisions

11.2 WORKSHEETS

PHASE: All LEM Phases

EQUIPMENT/ FUNCTION	*FAILURE MODE*	*UNCOMPENSATED FAILURE EFFECT*	*MISSION COMPLETION MODE*	*MISSION ABORT MODE*	*REMARKS*
Umbilical Hose-72" valved or 52" hose	Cut or punctured hose, manual control valve failed to open (72" only), damaged umbilical coupling	Loss of ability to provide ventilation, thermal control and humidity control in a pressurized cabin. Loss of life in a depressurized cabin environment	Prior to LEM separation: Replace failed hose with CM 42" standby hose. Commander uses remaining 52" or 72" hose (1). Separation to touchdown and ascent to docking: One crewman uses operable hose, the other has suit open or removed if body heat becomes excessive. Lunar exploration: Crewman in LEM uses operable umbilical hose. LEM cabin is to be repressurized as quickly as possible after extra-vehicular crewman returns. Extra-vehicular crewman remains connected to PLSS until cabin reaches a minimum of 3.5 psia. Thereafter, procedure applies as noted above for all other LEM phases (2) (3).	Prior to LEM separation: Perform normal abort. Separation to touchdown and ascent to docking: Perform normal abort to return to CM as soon as possible. One crewman uses operable hose; the other has suit open or removed if body heat becomes excessive. Lunar exploration: Same as for mission completion mode.	To optimize the useful exchange of CM and LEM hoses, the 42" CM standby hose and the 52" LEM hose should be lengthened, possibly to 52" and 72" respectively. (1) Present indications, based on LEM simulation studies, are that shorter hose lengths seriously restrict crew mobility within LEM. (2) A study should be made to determine the manner in which the PLSS of the extra-vehicular crewman should be used during the interval from return to LEM to the point where the PGA helmet visor can be initially opened. Study should include the need for purge flow during repressurization, and shut off of fan power to prevent possible motor burn out at higher than rated operating pressure levels. (3) Hose repair potential should be studied.

SUBSYSTEM: Crew Provisions

PHASE: All LEM Phases

EQUIPMENT/ FUNCTION	*FAILURE MODE*	*UNCOMPENSATED FAILURE EFFECT*	*MISSION COMPLETION MODE*	*MISSION ABORT MODE*	*REMARKS*
Electroluminescent (EL) Lighting, Control Panels and Displays	Faulty or shorted lamp connection, or usable brightness reduced to minimum, or EL circuit breaker fails open, or dimmer control fails to zero	Loss or reduction of legend visibility	Actuate backup flood lights in darkened panel area, and continue	Same as mission completion mode. Return to CM.	Present design provides EL lighting for all panel mounted instruments and displays with exception of signal lights and self-luminescent devices. Repairs are not contemplated in the light of system redundancies. If EL lamp or dimmer control fails to minimum brightness, deactivate appropriate circuit breaker.
Incandescent Lighting, Signal	Failure of single filament lamp, or failure of single filament in a double filament lamp	Reduction of brightness resulting in reduction of legend visibility	Continue normal mission		Incandescent lamps are used in multiples of two. Each single filament lamp is used in conjunction with a second lamp and will provide adequate illumination if one lamp fails. Each dual filament lamp will provide sufficient illumination with the remaining non-failed filament. The master warning signals (one group for each crewman) are illuminated by multiple lamps, and are accompanied by a simultaneous audio tone. Further study is recommended to determine the need for one, two, or no audio signal generators, in conjunction with these multiple lamps.
	Circuit breaker fails open	Loss of legend visibility			Failed open circuit breaker backup thru 2nd circuit breaker to 2nd survival buss, with no illumination break or dimming.
	Dimmer control fails	Loss or reduction of legend visibility			Dimmer by-pass switch returns signal lights to full illumination level, but dimming capability is lost. If dimmer light fails to minimum brightness manually, deactivate appropriate circuit breaker.
Incandescent Lighting, General (Dome Light Assemblies, two)	Faulty or shorted lamp, (one per assembly)	Reduced cabin illumination			Since both light assemblies have multi-directional capabilities, second dome light can provide adequate general illumination
Self-luminous devices	Luminescent source destroyed	Loss of capability to determine location or orientation of device	Actuate backup flood lights and continue		

Crew Provisions (Cont.)

11.3 DESCRIPTION

Crew Provisions are those crew supporting functions which are necessary for a successful LEM mission. In part, they include:

- Crew Support and Restraint
- Ingress-Egress Provisions
- Waste Management Provisions
- Space Suit Assembly Interface
- Interior Lighting
- Exterior Lighting

A description for Crew Support and Restraint is not included at this time since these provisions are in the development stage.

Ingress-Egress provisions include the forward and upper LEM hatches, the forward strut-mounted ladder and the forward hatch boarding platform. A discussion of the LEM hatches and the results of a contingency analysis performed on them is included in the Structural/Mechanical discussion, Section 8.

Waste Management provisions are discussed within the Space Suit Assembly description, Section 12.3, since a major portion of the system involves functions performed within the PGA of the SSA.

11.3.1 Space Suit Assembly Interface

Interfacing of the Space Suit Assembly and CM or LEM ECS is accomplished through vehicle umbilical hoses. Four hose assemblies, of varying lengths, are stowed in the CM at earth-launch, - with each assembly consisting of two flexible hoses and common couplings at each end. When used in a pressurized cabin, the umbilicals provide a gas transport between the ECS and the PGA's to implement PGA ventilation, and thermal and humidity control. In an unpressurized cabin, PGA pressurization may also be accomplished, in addition to those environmental features indicated above.

LEM vehicle umbilicals are 52" and 72" long and both are transferred to LEM prior to lunar orbit separation. The longer of the two umbilicals presently includes a manually operated shut-off valve, which was originally included to permit parallel interconnection of crewmen's PGA's during a "buddy" rescue. Since present design capabilities of the PLSS may not allow this mode of operation, the 72" umbilical shut-off valve may eventually be removed. Both LEM umbilical assemblies are further provided with electric cabling to establish a communications link between intra-vehicular crewmen and to telemeter each crewman's PGA biomedical sensor data.

11.3.2 Interior Lighting

The LEM interior lighting consists of panel, backup, and general illumination. Panel illumination includes integral or electroluminescent (EL) lighting which originates within a display component or indicator. Panel component illumination also includes signal lighting, - is accomplished by incandescent lamps located within the component and conveys intelligence information generated thru the requirements of other subsystems.

Backup capabilities for EL lighting is provided by activating local flood lights in the darkened panels. Signal light failure backup is provided by including multiple incandescent lamps within the signal component or by a second filament within each lamp.

General cabin lighting is provided by extendable, detachable, multiple dome light assemblies. Each assembly acts as backup for the other. A further provisional backup is contemplated through a self-contained, portable light source.

Legends and indicia pertinent to each instrument panel, or component within that panel, are acceptably discernible during all modes of operation described above.

11.3.3 Exterior Lighting

LEM exterior lighting presently includes multiple flood lights for use during docking operations. The lights are used to illuminate the CM area adjacent to the docking target and tunnel. Similar lights are incorporated on the CM for use during transposition docking. It is presently assumed that EL lighting will be used to illuminate the docking targets on both the CM and LEM though this has not been finalized. The possible use of the LEM dome lights as a backup for a docking target lighting failure is also being studied.

SECTION 12

SPACE SUIT ASSEMBLY

12.1 SUMMARY

Space Suit Assembly contingency analysis shows that most failures within this system will allow "mission continue" modes. In the majority of failure categories however, the "mission continue" modes must be modified to reschedule mission time lines and reduce lunar exploration. One such failure category involves malfunctions of non-interchangeable subsystems, thus limiting lunar exploration to one specific crewman. Such subsystems are the Pressure Garment Assembly (PGA) and the Liquid Cooled Garment (LCG) - with additional limiting possibilities in the External Thermal Garment (ETG) and the Meteoroid Protective Garment (MPG).

A second failure category permits interchangeability of failed and non-failed subsystems between crewmen, and will allow either crewman to perform an extra-vehicular mission, but again with reduced lunar exploration time. Such subsystems are the Portable Life Support System (PLSS) and Emergency Oxygen System (EOS).

Both above categories, however, create conditions that would preclude "buddy" rescue assistance in the event subsequent missions were to be undertaken.

One basic type of failure causing loss of life during lunar exploration, is a massive leak in a non-isolatable area of the PGA, - which because of its severity would either exceed make-up pressurization capabilities of the PLSS and EOS, or result in expending available O_2 before the exploring crewman's return to LEM could be accomplished. Meteoroidal penetration is a further possible cause for loss of life, with probable catastrophic results. The probability of PGA pressure vessel failure from external causes should be minimal, however, because of the multi-layered protection afforded the crewman by the ETG and MPG.

Loss of life could also be a possibility during the lunar exploration phase in the event a massive leak failure occurred in the PGA of the standby crewman in LEM while the other crewman is outside the spacecraft. If the leak were non-isolatable or not repairable, the difficultly might be overcome by the crewman in the failed PGA utilizing the combined make-up capabilities of LEM ECS and the intra-vehicular PLSS and EOS.

A number of Space Suit Assembly failures can cause mission abort. All involve PGA failures and are of such a nature as to jeopardize the life of the crewman should the LEM cabin be depressurized after the failure has occurred. They involve the following:

- Inability to maintain PGA pressurization e. g. : leaks, damaged seals and connectors, failed open PGA relief valve, etc.
- Inability to provide ventilation for PGA cooling and CO_2 removal while in a depressurized LEM cabin (blocked or short circuited helmet vent ducts).
- Inability of crewman to perform necessary intra-vehicular mission tasks in a depressurized environment because of PGA mobility or visibility failures. e. g: failed PGA restraint cables that would immobilize the legs or arms, - or would cause excessive helmet rise resulting in impaired vision; excessive abrasion to the helmet primary visor resulting in unacceptable optical characteristics.
- Inability of the crewman to use the PGA waste management system while suited. e.g: internally leaking urine collection and/or transfer system.

A final category of Space Suit Assembly failures will, in most instances, require physical "buddy" assistance by the standby crewman in LEM if they occur during lunar exploration. These failures again involve only the PGA and are as follows:

- Failures causing excessive PGA mobility decrement (restraint cable failures primarily affecting the walking mode).
- Failures causing loss of vision (dust contamination of external surfaces; contamination of visor internal surfaces by coughing, sneezing, etc; temporary blindness caused by failure of anti-glare visor).

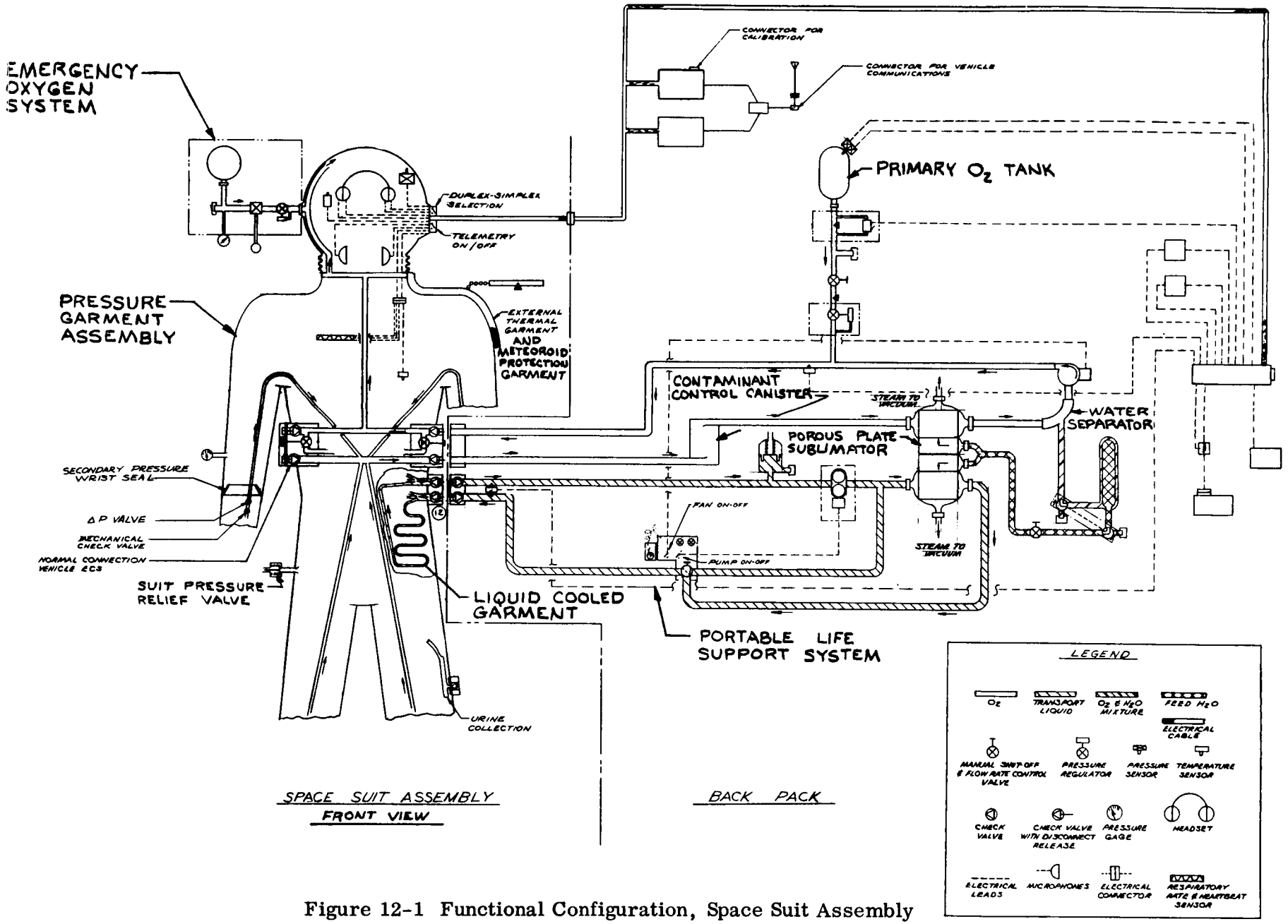


Figure 12-1 Functional Configuration, Space Suit Assembly

SUBSYSTEM: Space Suit Assembly

12.2 WORKSHEETS

PHASE: Lunar Exploration

EQUIPMENT/ FUNCTION	*FAILURE MODE*	*UNCOMPENSATED FAILURE EFFECT*	*MISSION COMPLETION MODE*	*MISSION ABORT MODE*	*REMARKS*
PLSS No. 1 or No. 2	Inoperative	Loss of life	Return to LEM using emergency backup capability provided. Switch to non-failed PLSS for continued exploration.	Begin normal abort procedure at first opportunity.	One inoperative PLSS precludes possibility of "buddy" rescue for subsequent missions. If failure of PLSS prevents use of liquid coolant garment under space suit (such as by draining transport fluid through leak in PLSS) then that crewman would be prevented from performing any subsequent EV mission. Repair and/or replacement possibilities for failed component should be studied.
PLSS pressurization	Failed closed primary regulator		Actuate EOS and return to LEM. Switch to non-failed PLSS for continued exploration.		Failure detected by audible low suit pressure warning sensor. EOS capable of duplicating function of primary oxygen system for 30 minutes providing otherwise normal PLSS operation.
	Failed open primary regulator		Return to LEM at first indication. Actuate EOS only after low suit pressure indication. Switch to non-failed PLSS for continued exploration.		Failure detected by audible high primary O_2 flow rate sensor. Suit pressure will be limited by suit relief valve setting. EOS would not have to be actuated until audible low pressure warning signal is given. EOS capability slightly augmented by primary O_2
	Excessive leakage (of PLSS)		Same as above		Failure detected audibly with high O_2 flow rate sensor and/or low suit pressure sensor. Failure also may be detected by self-readout and/or telemetry of O_2 primary quantity. Available time to rescue is a function of leak rate and quantity of primary O_2 to augment EOS. Parallel operation capabilities of EOS regulator and primary regulator should be studied.
PLSS ventilation	Loss of electrical power, fan inoperative, loss of O_2 heat sink	Loss of life due to CO_2 build up, Loss of humidity control, Loss of limited cooling capability of vent gas	Actuate EOS and helmet purge valve and return to LEM. Switch to non-failed PLSS for continued exploration.		Purge of CO_2 requires flush-through operation. Duration of O_2 supply for CO_2 purge function of quantity of primary O_2 to augment EOS. If ventilation failure caused by battery failure, no cooling of any type provided by PLSS. If heat sink is lost, electrical circuit to fan/motor must be broken to prevent excessive temperature rise of ventilation gas. If malfunction due to failed battery, a spare battery (if available) can restore failed PLSS to operational status.
PLSS thermal control	Loss of expendable water supply	Complete loss of cooling capability with excessive sweating and possible dehydration. Eventual loss of life due to body overheat	Return to LEM at first indication of loss of water or overheat. Disconnect electrical power to fan when inlet O_2 temperature becomes excessively high. Actuate EOS and helmet purge valve. Disconnect electrical power to pump when liquid coolant temperature becomes excessively high. Switch to non-failed PLSS for continued exploration.		If water loss is due to leakage, failure can be initially detected by feed back from reservior H_2O quantity indicator. If water loss is due to water blockage and/or heat exchanger malfunction, failure can be detected by O_2 inlet temperature indication (telemetry). Overheat of inlet O_2 and liquid transport cooling fluid forces shut down of recirculation systems resulting in loss of ventilation, humidity control and contaminant control functions and necessitates subsequent EOS and purge valve actuation.
	Loss of O_2 to expendable water heat exchanger (HX)	Loss of head and respiration cooling	Return to LEM at first indication of overheat. Disconnect electrical power to fan when inlet O_2 temperature becomes excessively high. Actuate EOS and helmet purge valve. Switch to non-failed PLSS for continued exploration.		Overheat of inlet O_2 forces shut down of gas recirculation system resulting in loss of ventilation, humidity control and contamination control functions and necessitates subsequent EOS and purge valve actuation.

SUBSYSTEM: Space Suit Assembly

PHASE: Lunar Exploration

EQUIPMENT/ FUNCTION	*FAILURE MODE*	*UNCOMPENSATED FAILURE EFFECT*	*MISSION COMPLETION MODE*	*MISSION ABORT MODE*	*REMARKS*
PLSS thermal control	Loss of liquid transport to expendable water heat exchanger (HX).	Body overheat due to loss of primary limb and torso cooling source with excessive sweating and possible dehydration	Return to LEM at first indication of liquid coolant overheat. Switch to non-failed PLSS for continued exploration.	Begin normal abort procedure at first opportunity.	Limited cooling of body possible by O_2 ventilation loop plus heat storage capability of liquid loop subsystem. Failure detection possible if liquid coolant temperature is telemetered.
	Leakage of liquid transport cooling fluid	Same as above	Return to LEM at first indication of liquid coolant overheat. Further exploration may be done by second crewman using non-failed PLSS.		Limited cooling of body possible by O_2 ventilation loop. Failure detection possible if liquid coolant temperature is telemetered. The potential for recharging the liquid cooled garment from LEM water supply should be studied to enable either crewman to perform subsequent exploration.
	Liquid transport pump failure	Same as above	Return to LEM at first indication of liquid coolant over heat or pump failure Switch to non-failed PLSS for continued exploration.		Limited cooling of body possible by O_2 ventilation loop plus heat storage capability of liquid loop subsystem. Failure detection possible if liquid coolant temperature is telemetered. Pump failure detection possible by telemetered indication of battery current.
	Temp control valve failure to maximum cooling	Body sub-cooling at low metabolic rates	Reprogram activity level to acceptable subjective comfort level and continue mission. Switch to non-failed PLSS for continued exploration.		Mission duration shortened by work at higher than programmed average rate. Certain "low activity" tasks may have to be eliminated.
	Temp control valve failure to minimum cooling	Body overheat due to reduced cooling to limbs and torso with excessive sweating and possible dehydration	Return to LEM at first indication of liquid coolant overheat. Switch to non-failed PLSS for continued exploration.		
PLSS humidity control	Loss of water separator performance	Possible impairment to vision and/or discomfort due to free moisture introduced to helmet, torso and limbs	Return to LEM at first evidence of visual impairment. Switch to non-failed PLSS for continued exploration.		
PLSS contamination control	Loss of CO_2 removal capability	Loss of life due to CO_2 buildup	Actuate EOS and helmet purge valve and return to LEM.		Purge of CO_2 requires flush-through operation. Duration of O_2 supply for CO_2 purge function of quantity of primary O_2 to augment EOS. If malfunction due to failed contaminant control cartridge, a replacement cartridge (if available) can restore PLSS to operational status. Study should be made to determine if supplied physiological sensors can provide an indication of CO_2 build up in time to permit crewman to take contingent action.
	Loss of particulate filtration capability	Potential lung damage due to possible migration of LiOH particles	Return to LEM at first indication. If contamination to oral-nasal area severe, disconnect electrical power to fan and actuate EOS and purge valve. Switch to non-failed PLSS for continued exploration.		If residual contamination level in helmet vent line is severe, subsequent vehicle ECS purge with visor open may be required.

SUBSYSTEM: Space Suit Assembly

PHASE: Lunar Exploration

EQUIPMENT/ FUNCTION	*FAILURE MODE*	*UNCOMPENSATED FAILURE EFFECT*	*MISSION COMPLETION MODE*	*MISSION ABORT MODE*	*REMARKS*
PLSS contamination control	Loss of noxious odor removal capability	Nausea and discomfort	Return to LEM at first indication. If odor condition severe, intermittently operate helmet purge valve to flush. Replace contaminant control cartridge (if available) to permit PLSS use for continued exploration.	Begin normal abort at first opportunity.	
PLSS breathing oxygen	Blockage of O_2 make-up line	Loss of life	See comments for failed closed primary regulator under "pressurization function."		See remarks for failed closed primary regulator.
	Leakage of O_2 make-up line	Same as above	See comments for excessive leakage of PLSS under "pressurization function."		See remarks for excessive leakage of PLSS.

SUBSYSTEM: Space Suit Assembly

PHASE: Lunar Exploration

EQUIPMENT/ FUNCTION	*FAILURE MODE*	*UNCOMPENSATED FAILURE EFFECT*	*MISSION COMPLETION MODE*	*MISSION ABORT MODE*	*REMARKS*
PGA pressurization	Excessive leakage due to structural failure or failure of mechanical seals	Can range from reduction of mission time to loss of life due to loss of pressure	See remarks	(1) Initiate return to LEM at first indication of malfunction. (2) Actuate EOS only after low suit pressure indication. (3) Reconnect to LEM ECS & repressurize cabin as quickly as possible. (4) Continue with normal abort procedure to return to CSM as soon as possible.	(1) Excessive PGA leakage could preclude any subsequent extra vehicular mission operation depending on magnitude of leakage and adaptability to repair. High leak rate could preclude any rescue possibility. (2) Possibilities exist for "Patch" repair of damaged area of PGA or seal replacement if leakage source can be located. A suit repair kit is contemplated as in flight spare equipment. Repair may allow normal continuation of mission. (3) Means of accurately determining PGA leakage rate during subsequent intra-vehicular checkout should be studied to determine future "Use" limitations of damaged PGA. (4) Ability to maintain PGA pressure in event of subsequent loss of cabin pressure may be compromised. (5) Remarks under PLSS pressurization applicable to "Excessive Leakage" failure mode also apply.
	Failed open PGA relief valve	Loss of life	No further exploration possible by crewman in failed PGA. See remark (6)	Comments (1) through (4) above apply.	Remarks (4) and (5) above apply. (6) Repair potential should be studied. Further exploration possible by second crewman if LEM ECS can make up leak rate.
	Failure of glove	Can range from hand extremity damage to loss of life	Return to LEM at first indication Actuate EOS if low suit pressure indication occurs. Reconnect to LEM ECS and repressurize cabin as quickly as possible. If no physiological damage occurs, a switch to a spare glove or E. V. glove of a 2nd crewman and subsequent continuation of mission is possible	Comments (1) through (3) above apply. Begin normal abort procedure at first opportunity.	(7) PGA is designed such that in event of glove leak or tear, a secondary wrist seal will be actuated to maintain maintain suit pressurization. Because of greater susceptibility of gloves to damage during E. V. mission, spare requirements for gloves should be studied with consideration to the possibility of a single size accommodating both crewmen.
	Perforation by meteoroid	Probable loss of life due to loss of pressure if loss of life is not otherwise caused by physiological damage resulting from explosion or effects of body penetration	See remarks (2) and (3)	Comments (1) through (4) above apply if no incapacitating physiological damage is suffered.	Remarks (1) through (5) above apply.
PGA ventilation	Constriction or blocking of vent distribution ducts	Reduction or loss of head cooling Reduction or loss of CO_2 removal capability with loss of life in the extreme due to excessive CO_2 build up Possible loss of vision for lunar night mission or day shadow exploration due to visor frosting or fogging	See remarks	Actuate EOS and helmet purge valve and return to LEM. Comment (3) above applies. Open helmet visor as soon as safe level of cabin pressure is reached. Comment (4) above applies.	(8) Purge of CO_2 requires flush through operation. (9) Duration of O_2 supply for CO_2 purge function of quantity of primary O_2 to augment EOS, if blockage is not in helmet inlet line ducting. (10) Constriction of vent ducts might be inferred from indicated rise in suit inlet temperature coupled with EKG indication. (11) Excessive rise of suit inlet temperature (as measured in PLSS ducting) will require shut down of power to PLSS recirculating fan. (12) Helmet purge valve should not be removed until visor is open unless it can be established that constriction is not part of distribution system associated with ECS operation.

SUBSYSTEM: Space Suit Assembly

PHASE: Lunar Exploration

EQUIPMENT/ FUNCTION	*FAILURE MODE*	*UNCOMPENSATED FAILURE EFFECT*	*MISSION COMPLETION MODE*	*MISSION ABORT MODE*	*REMARKS*
PGA ventilation (Cont)	Internal duct leak causing short circuit of vent flow	Same as for constriction or blockage of distribution ducts	See remarks	Same as for failure due to constriction or blocking of vent distribution ducts.	(13) If duct failure is in helmet inlet line, subsequent cabin depressurization for additional exploration missions is not possible. If duct failure is elsewhere, further exploration by second crewman is possible if crewman in failed PGA switches ECS hose to PGA connectors on opposite side of torso. Remarks (8), (9) and (13) above apply. Helmet purge valve should not be removed until visor opened.
	Failed open vent flow control valve	Reduction in head cooling, Reduction in CO_2 removal capability to potentially dangerous levels if high work rate maintained	Return to LEM at first indication Further exploration may be done by 2nd crewman	Begin normal abort procedure at first opportunity.	Vent flow control valve normally closed for extra vehicular operation. Opening of valve will have effect of by-passing part of ventilation flow to helmet. Failed open valve precludes possibility of PGA use for "buddy" rescue for subsequent missions.
PGA mobility	Failure of shoulder or elbow convolute restraint cable	Loss or severe reduction of use of affected arm due to over extension of affected convolute joint	See remarks	Return to LEM at first indication of malfunction. Begin normal abort procedure to return to CSM as soon as possible.	Loss of arm mobility may require "buddy" assistance to re-enter LEM, hook up to ECS and perform other duties prior to cabin repressurization.
	Failure of glove restraint cable	Loss or severe reduction in use of affected hand	Return to LEM at first indication. A switch to a spare glove or E. V. glove of second crewman and subsequent continuation of mission is possible	Begin normal abort procedure at first opportunity.	Loss of hand mobility may require buddy assistance to re-enter LEM, hookup to ECS and perform other necessary functions prior to cabin repressurization. Also see remark (7) under PGA pressurization applicable to "Failure of Glove".
	Failure of crotch restraint cable, hip or knee convolute restraint cable, or torso length adjustment strap	Loss or severe reduction in walking capability and leg mobility due to extension and/or cocking of affected joint	See remarks	Return to LEM with "buddy" assistance if necessary. Begin normal abort procedure. Return to CSM as soon as possible.	Loss or reduction of leg mobility will most likely require "buddy" assistance to enable crewman to return to LEM and perform necessary functions prior to cabin repressurization. See also remark (14)
	Failure of boot holddown adjustment strap	Loss or severe reduction in walking capability	See remarks	Same as above.	Same as above. (14) Possibilities exist for subsequent exploration by crewman in non-failed PGA if necessary tasks in LEM depressurized cabin can be performed by crewman in failed PGA.
	Failure of helmet convolute restraint and pivot mechanism	Loss or severe reduction in helmet nodding capability and restriction to visibility due to helmet rise	See remarks	Same as above.	Restriction in field of vision will probably require "buddy" assistance to perform necessary functions prior to cabin repressurization. See also remark (14)
	Failure of front torso adjustment device (cable and pulley system)	Loss or severe reduction in bending capability	See remarks	Same as above.	See also remark (14)

SUBSYSTEM: Space Suit Assembly

PHASE: Lunar Exploration

EQUIPMENT/ FUNCTION	*FAILURE MODE*	*UNCOMPENSATED FAILURE EFFECT*	*MISSION COMPLETION MODE*	*MISSION ABORT MODE*	*REMARKS*
PGA visibility	Failure of helmet convolute restraint	See comments under same failure mode for PGA mobility		Return to LEM with "buddy" assistance if necessary. Begin normal abort procedure. Return to CSM as soon as possible.	See comment under same failure mode for PGA mobility.
	Failure of neck ring holddown cable	Reduction of visibility due to helmet rise		Same as above.	
	Visor abrasion and/or dust coating	Reduction and/or loss of visibility	Return to LEM at first indication of degraded vision with "buddy" assistance if necessary. If permanent damage has been done to optical characteristics of supplemental visor or visors, subsequent continuation of mission is possible by switching to supplemental visor assembly of 2nd crewman.	Begin normal abort procedure at first opportunity.	
	Visor fogging due to failure of vent system and/or visor antifog coatings	Same as above	No continue mode if PGA vent failure is cause. If visor coating is the cause, then mission may be continued by 2nd crewman.	If cause is due to vent failure see comments for constriction or blocking of vent distribution ducts failure mode. Otherwise begin abort procedure at first opportunity.	If visor fogging occurred during lunar day exploration such as when going into shade, reorientation of visor to see reflected solar radiation could provide some defogging assistance.
	Contamination of visor from internal sources as coughing, sneezing, etc.	Reduction of visibility	Return to LEM at first indication. Clean visor internal surfaces after cabin repressurization. Exploration can then be continued normally.	Begin normal abort procedure at first opportunity.	Study should be made to establish what methods are best suited for visor cleaning when inside LEM.
	Inoperable-anti glare visor in "non-use" position	Possible eye damage and impaired vision, Inability to see contrast due to glare.	Return to LEM at first indication. Continuation of mission is possible by switching to supplemental visor assembly of 2nd crewman.	If no eye damage causing physical or visual impairment, begin normal abort procedure at first opportunity. Otherwise, return to CSM as soon as possible.	If glare condition is severe, "buddy" assistance may be required to permit return to LEM.
PGA waste management	Internal leakage of urine management system	Discomfort due to wetting and odors	See remarks.	Return to LEM at first indication. If odor condition severe, intermittently operate helmet purge valve to flush. Begin normal abort procedure. Return to CSM as soon as possible.	Possibilities of repair to urine management system in LEM pressurized cabin environment should be studied for the possibility of allowing subsequent continuation of mission.
PGA supplemental equipment restraint or supports	Failure of PLSS attachment point on PGA	Decrement to walking capability due to loose attachment of PLSS.	Return to LEM at first indication. Continuation of mission is possible by second crewman using either PLSS #1 or #2.	Begin normal abort procedure at first opportunity.	
LCG heat removal capability	Obstruction of main distribution ducting in LCG	Body overheat due to loss of primary limb and torso cooling, with excessive sweating and possible dehydration	Return to LEM at first indication of liquid coolant overheat. Further exploration may be done by 2nd crewman.	Same as above.	Limited body cooling possible by O_2 ventilation loop. Malfunction of LCG precludes possibility of "buddy" rescue for subsequent missions.
	Crimping or obstruction of local distribution ducting	Localized loss of cooling in area of obstruction	Same as above.		Same as above.
	Leakage of liquid transport cooling fluid from LCG	Body overheat due to loss of primary limb and torso cooling, with excessive sweating and possible dehydration	Return to LEM at first indication of body overheat. Further exploration may be done by second crewman.		Same as above.

SUBSYSTEM: Space Suit Assembly

PHASE: Lunar Exploration

EQUIPMENT/ FUNCTION	*FAILURE MODE*	*UNCOMPENSATED FAILURE EFFECT*	*MISSION COMPLETION MODE*	*MISSION ABORT MODE*	*REMARKS*
ETG thermal protection	Garment tear. Degradation of surface thermal optical properties due to dust adhesion or abrasion	Possible body overheat during lunar day mission, or body subcooling during lunar night mission, due to higher than programmed heat leakage to or from the crewman.	Return to LEM at first indication of physical damage or overheat. Further exploration may be done by second crewman.	Begin normal abort procedure at first opportunity.	If degradation of ETG performance caused by tear with known location, reorientation of suit to see a less severe solar radiation view factor is possible to minimize heat leak in during lunar day mission. A study should be made to determine the possibility of a universal size ETG to fit all crewmen.
MPG meteoroid protection	Damage or failure of local section of MPG to cover body area	Increased potential to meteoroid penetration of PGA (see "perforation by meteoroid" failure mode under PGA pressurization function for effects)	Return to LEM at first indication of physical damage. Further exploration may be done by second crewman.	Same as above.	A study should be made to determine the possibility of a universal size MPG to fit all crewmen.

SUBSYSTEM: Space Suit Assembly

PHASE: All LEM Phases

EQUIPMENT/ FUNCTION	*FAILURE MODE*	*UNCOMPENSATED FAILURE EFFECT*	*MISSION COMPLETION MODE*	*MISSION ABORT MODE*	*REMARKS*
EOS #1 or #2 (emergency pressurization and breathing oxygen supply for PGA)	Inoperative due to external leakage, failed open regulator (2) or failed closed regulator.	None, if no PLSS failure (1)	Continue normal mission but operable EOS remains with crewman performing lunar exploration.	Perform nominal abort.	(1) Failure of EOS is significant only when double failures are considered, e.g., after a previous failure of PLSS. Since the EOS performs as the only emergency pressurization and breathing O2 backup source for the PLSS, the EOS expiration rates should be studied when determining lunar exploration time limits. Study also the advisability of returning EOS' to the CM for use during a possible EVT. (2) Open regulator concurrent with EOS actuation (double failure) will overpressurize the PGA and cause probable PGA structural failure.
	Burst disk shut-off valve fails to open.	Delayed suit depressurization upon re-entry into LEM from EVO (1)	Same as above.	Same as above.	(1) Flow rates of the EOS are controlled by a Δ P type pressure regulator. After the crewman re-enters LEM, his ability to depressurize his PGA will be delayed several minutes until the EOS has been exhausted into the cabin.

SUBSYSTEM: Space Suit Assembly

PHASE: All LEM Phases Except Lunar Exploration

EQUIPMENT/ FUNCTION	*FAILURE MODE*	*UNCOMPENSATED FAILURE EFFECT*	*MISSION COMPLETION MODE*	*MISSION ABORT MODE*	*REMARKS*
PLSS pressurization	Failed open primary regulator, Failed closed primary regulator, Damaged umbilical coupling, Excessive leakage of PLSS	None If leak from high pressure O_2 supply, slight cabin pressure buildup may occur.	Preparation for LEM separation to docking. Continue nominal mission (1). Mission time-line rescheduled to permit single PLSS operation (3).	All LEM phases except lunar exploration: Perform normal abort (2).	Except for the lunar exploration phase, the PLSS has backup or potential functions only. None of the failures considered make a successful abort impossible. Donning of PLSS as ECS backup during descent or ascent phases not advisable because of possible biaxial loads encountered at lunar landing and liftoff. (1) PLSS-PGA pressure integrity should be checked before starting major portion of LEM preseparation checkout. If third PLSS was available as backup, it would make interchangeability of crewmen and associated equipment possible between LEM and CM. Study should be made of above interchange in view of possible additional crew training and weight penalties. (2) Analysis should be performed to evaluate need for return of operative PLSS to CM, if mission aborted. (3) Analysis of lunar operations with a single PLSS and crew rotation should be performed. One inoperative PLSS precludes possibility of lunar surface "buddy" rescue, precludes intravehicular ECS backup for one crewman, precludes extra-vehicular return of one crewman in the event of no hard dock and probably reduces lunar EV mission time. These restrictions should be considered before attempting continuation of the mission with one PLSS. (4) Effect of free water on cabin equipment and ECS should be studied.
PLSS ventilation	Loss of electrical power, Fan inoperative, Loss of O_2 heat sink	None			
PLSS thermal control	Loss of expendable H_2O supply	Potential leak of free water into cabin (4).			
	Leakage of liquid transport cooling fluid	Same as above			
	Loss of O_2 to expendable H_2O HX, Loss of liquid transport to expendable H_2O HX, Liquid transport pump failure, Temp control valve failure to max. cooling, Temp control valve failure to min. cooling	None			
PLSS humidity control	Loss of H_2O separator performance	None			
PLSS contamination control	Loss of CO_2 removal capability, Loss of particulate filtration capability, Loss of noxious odor removal capability	None			Loss of contaminant control canister from PLSS removes a potential backup source for ECS.
PLSS breathing oxygen	Blockage of O_2 make-up	None			
	Leakage of O_2 make-up lines	See "Excessive leakage of PLSS" under PLSS pressurization function.			

SUBSYSTEM: Space Suit Assembly

PHASE: All LEM Phases Except Lunar Exploration

EQUIPMENT/ FUNCTION	*FAILURE MODE*	*UNCOMPENSATED FAILURE EFFECT*	*MISSION COMPLETION MODE*	*MISSION ABORT MODE*	*REMARKS*
PGA pressurization	Excessive leakage due to structural failure, or failure of mechanical seals.	Loss of capability to maintain steady-state suit pressure with consequent mission abort	Prior to LEM separation:- Crewman with damaged PGA could assume the CM duties during all further mission phases (1). Separation to Descent Initiation:- Redock, exchange crewmen, continue. Descent Initiation to Docking:- See remarks (2).	All LEM phases except lunar stay:- Begin normal abort to return to CM as soon as possible. Lunar stay:- Combined outputs of LEM, EOS and PLSS (primary O_2) may be used to overcome low PGA pressure of crewman in LEM during an emergency ingress. If not sufficient, one crewman loses life.	In all phases but lunar stay, the PGA has backup functions only. None of the failures considered make a successful abort impossible. (1) The ability to exchange crewmen is necessary for mission completion because steady-state PGA pressurization is mandatory for both PGAs when the LEM cabin is depressurized for lunar exploration. Since the PGA's, LCG's, ETG's and MPG's are "tailored" to each crewman, they are not usually interchangeable. This proposes the advisability of all crewmen receiving equal mission training, if crewmen ability and weight penalties permit. Because the CM crewman does not have an LCG, ETG or MPG, the mission must be accomplished with only one crewman extra-vehicular. At present, an LCG, ETG and MPG for the CM crewman would be required to have crew interchangeability without reducing LEM mission capabilities. Crew selection, as to size, should be studied to determine the feasibility of LCG, ETG and MPG interchangeability between crewmen. (2) Feasibility of continuing mission depends on extent or degree of repair capability
	LH umbilical coupling damaged	Loss of capability to provide PGA pressurization thru PLSS umbilical	Same as above. See remarks (3).	Perform normal abort.	(3) Crew rotation for lunar exploration will not be possible. Study of extending length of PLSS umbilical hoses to enable coupling to either side of PGA should be considered.
	Pressure relief valve failed open	Loss of capability to maintain steady-state PGA pressure	Same as above. See remarks (4).	Same as above.	(4) Repair potential should be studied. High leak rate could preclude any rescue possibility.
	Failure of glove	Can range from extremity damage to loss of life	Use spare glove and continue.	Same as above	PGA is so designed that in the event of a glove leak or tear, a secondary wrist seal will actuate and maintain suit pressure.
PGA ventilation	Constriction or blocking of vent distribution ducts Internal duct leak causing short circuit of vent flow	Reduction in head cooling and CO_2 removal capability	Prior to LEM separation:- Crewman with damaged PGA could assume the CM duties during all further mission phases (1). Separation to Descent Initiation:- Redock, exchange crewmen, continue. Descent Initiation to Docking:- See remarks (1) (2).	Same as above.	See remarks (13) under PGA pressurization failure during lunar exploration phase.

SUBSYSTEM: Space Suit Assembly

PHASE: All LEM Phases Except Lunar Exploration

EQUIPMENT/ FUNCTION	*FAILURE MODE*	*UNCOMPENSATED FAILURE EFFECT*	*MISSION COMPLETION MODE*	*MISSION ABORT MODE*	*REMARKS*
PGA ventilation	Vent flow control valve failed open	Reduction in head cooling and CO_2 removal capability by ECS due to helmet bypass.	Exchange ECS umbilical hose point of attachment to PGA failed coupling and continue. Exploration may be done by 2nd crewman.	Perform normal abort.	Crew rotation for lunar exploration will not be possible, consequently precludes possibility of lunar surface rescue.
LCG heat removal capability	Obstruction of main distribution ducting or leakage of transport fluid from LCG	Loss of capability for PGA heat removal during lunar exploration	Prior to LEM separation:- Crewman with damaged LCG could assume the CM duties during all further mission phases (1). Separation to Descent Initiation:- Redock, exchange crewmen, continue. Descent Initiation to Docking:- See Remarks		Same as above.
	Localized crimping or obstruction in ducting	Loss of local cooling			
PGA mobility	Failure of restraint or adjustment devices	Loss, or severe reduction of use of affected areas	Prior to LEM separation:- Crewman with damaged PGA could assume the CM duties during all further mission phases (1). Descent Initiation to Docking:- Insertion to lunar egress:-		Restraint failures, depending on the severity of the mobility decrement, could preclude any subsequent EV operation because of inability of the crewman in the failed PGA to perform necessary intra-vehicular mission tasks in a depressurized cabin environment.
PGA visibility	Failure of helmet or neck ring restraint devices	Reduction, or loss of visibility	Same as above.		Same as above.
	Visor abrasion	Same as above	If abrasion is permanent, continuation of mission possible by interchange of visor, or visors (except pressurized visor) with 2nd crewman.		Crew rotation possible for lunar exploration, but degrades potential of lunar surface rescue.
	Visor internal contamination, - as from coughing, sneezing, etc.	Reduction of visibility	Clean visor, or visors to the degree possible, and continue.		The use of wiping devices should be considered and stowed in LEM.
PGA waste management system	Internal leakage of urine to PGA Internal constriction or blocking of PGA urine waste ducts	Discomfort due to wetting and odor Loss of urinating capability due to PGA waste system back pressure	Prior to LEM separation:- Crewman with damaged WMS could assume the CM duties during all further mission phases. Separation to Descent Initiation:- Redock, exchange crewmen, continue. Descent Initiation to Docking:- See remarks.		Study possibilities of repair to urine management system. A study should also be made to determine the best type of vehicle waste management backup provisions which can be used without a PGA WMS.

SUBSYSTEM: Space Suit Assembly

PHASE: All LEM Phases Except Lunar Exploration

EQUIPMENT FUNCTION	*FAILURE MODE*	*UNCOMPENSATED FAILURE EFFECT*	*MISSION COMPLETION MODE*	*MISSION ABORT MODE*	*REMARKS*
Supplemental equipment restraint or supports	Failure of PLSS attachment point on PGA	Decrement to walking capability due to loose attachment of PLSS	Prior to LEM separation: - Crewman with damaged PGA could assume the CM duties during all further mission phases. Separation to Descent Initiation: - Redock, exchange crewmen, continue. Descent Initiation to Docking: - See remarks.	Perform normal abort.	Crew rotation for lunar exploration will not be possible, - consequently precludes possibility of "buddy" rescue.
Thermal protection from external thermal garment (ETG)	Garment tear. Degradation of surface thermal optical properties due to dust adhesion or abrasion	Loss of capability to prevent body overheat during Lunar day mission or body subcooling during lunar night mission due to higher than programmed heat leakage to or from the crewman	Continuation of mission possible by 2nd crewman.		
Visual protection from supplemental helmet visors	Inoperable anti-glare visor in "non-use" position	Possible eye damage and impaired vision. Inability to see contrast due to glare.	Continuation of mission possible by interchange of supplemental visor assembly of 2nd crewman.		
Meteoroid protection from meteoroid protective garment (MPG)	Damage or failure of local section of MPG to cover body area	Loss of capability to provide protection from meteoroids	Continuation of mission possible by 2nd crewman.		

12.3 DESCRIPTION

The Space Suit Assembly provides life support when the astronaut is external to the LEM or CM cabin environment. It further provides a limited-time emergency life supporting backup mode should cabin pressure be lost during any phase of the mission. The Space Suit Assembly consists of six major subsystems: Portable Life Support System (PLSS); Pressure Garment Assembly (PGA); Liquid Cooled Garment (LCG); External Thermal Garment (ETG); Meteroid Protection Garment (MPG); Emergency Oxygen System (EOS). For intra-vehicular operation inside the command module, a constant wear garment (CWG) is worn under the PGA in lieu of the LCG. Additional subsystems integrated within the SSA are waste management, biomedical (EKG) and environmental sensors, as well as communications and telemetry devices. Accessory umbilical hoses provide additional emergency backup capability between SSA's.

12.3.1 Portable Life Support System (PLSS)

The PLSS is a portable, self-contained, rechargeable assembly providing a limited-time life support system for each crewman during extra-vehicular operations and unpressurized LEM operations associated with egress. One fully charged PLSS (used under nominal work load conditions) provides a three hour capability for lunar exploration with a possible small additional contingency time available. If used in free space operations, at reduced work load conditions, the PLSS can provide life support for a period up to four hours.

12.3.1.1 Pressurization: Normal PGA pressure control is provided through the PLSS primary O_2 supply and its pressure regulator at a rate that maintains the PGA at 3.7 psig. Additional emergency O_2 pressure is available by activating the EOS.

12.3.1.2 Ventilation: Ventilation of the PGA is accomplished by a fan operated, closed O_2 loop conditioning and circulating system.

12.3.1.3 Thermal Control: SSA thermal levels are primarily maintained by transfer of heat from a LCG transport fluid recirculation loop to a water sublimator heat exchanger which is supplied from an independent PLSS expendable water source. Limited cooling is also provided by the heat absorption properties of the O_2 circulating within the PGA and transferring its heat to a PLSS O_2 loop sublimator.

12.3.1.4 Humidity Control: Moisture is condensed in the PLSS O_2 loop sublimator and is removed by an H_2O separator. Separated water is stored within the PLSS for subsequent removal and transfer to the vehicle ECS waste water system during PLSS recharge.

12.3.1.5 Contamination Control: Control of CO_2 and noxious gases is maintained by passing all PGA atmosphere circulatory gases through a replaceable cartridge containing LiOH and activated charcoal.

12.3.1.6 Breathing O_2: Breathing O_2 is provided by a PLSS primary O_2 Reservoir which is rechargeable from space craft stores. An independent, non-rechargeable O_2 reservoir is supplied as part of the EOS for back-up purposes.

12.3.1.7 Electrical Power: SSA electrical power during extravehicular operations is supplied by rechargeable, silver zinc, 16.8V battery.

12.3.2 Pressure Garment Assembly (PGA)

The PGA is a flexible anthropomorphous pressure vessel completely enclosing the crewman and consists of a booted garment covering the torso and limbs but with detachable helmet, gloves and waste management components. The PGA is worn throughout all LEM operations. It provides the crewman with a ventilated environment in a pressurized LEM cabin and a pressurized and ventilated environment in a zero pressure environment. Fittings are included to allow coupling to the PLSS, LCG, and ECS, waste management and communications systems. Mobility of the PGA is such that the crewman will be able to walk, climb, crouch and rise from a supine to a standing position.

12.3.2.1 Helmet: The functional purpose of the helmet and its internal pressure visor is primarily to extend the PGA pressure vessel capabilities to the crewman's head, and secondarily to protect his head from impact load injury.

The helmet provides an extension of the PGA ventilation distribution system for respiration needs, head ventilation and visor defogging. There is an additional connection from this system to the EOS which is mounted on the helmet.

With the internal visor closed and the PGA pressurized, a valve appropriately located on the visor surface provides a port permitting the crewman to eat, drink, and in an emergency to purge his helmet of excess CO_2.

The helmet further contains redundantly wired communications assemblies, as well as audible warning devices for low PGA pressure and high PLSS primary O_2 flow. All electrical connections are made thru the helmet to an external pigtail. The wide range of illumination and environmental hazards on the lunar surface necessitates additional flip down visors attached to the helmet's exterior. These supplemental visors are detachable from the helmet.

12.3.2.2 Torso and Limb Suit: The torso and limb suit encloses the crewman's body and limbs, excluding the head and hands. Used in conjunction with the helmet and gloves, the torso and limb suit provides an enclosure within which a 3.7 psi atmosphere may be maintained. In addition, an integrated ventilation system has been included to circulate oxygen to fulfill the physiological requirements of ventilation and humidity control, thermal comfort, CO_2 removal and total PGA pressurization. It further incorporates provisions for attachment and routing of EKG and environmental instrumentation and for an internal waste management transfer system.

12.3.2.3 Gloves: PGA gloves will extend the PGA pressure vessel to the crewman's hands. They provide adequate finger dexterity while pressurized. A glove tear in a depressurized environment will cause the hand to be sealed off from the body by a cuff seal at the wrist.

12.3.2.4 Waste Management: In-suit urine management is performed by a penis seal and receptacle connected to a urine collection bag. Present volume requirements for this bag, assuming that the bag is not emptied during the LEM mission, is approximately 950cc. Such a volume may be difficult to contain within the PGA and the feasibility of external collection is being studied. Present design allows for an external quick disconnect on the PGA through which the bag may be drained, possibly into a container located within the LEM descent stage.

Additional design study is required before a final solution to fecal waste management is established. It is assumed that during a normal mission, defecation will not be required because of crew conditioning and low residue diets.

12.3.3 Liquid Cooled Garment (LCG)

The LCG is a liquid-cooled garment worn under the PGA during all LEM and extra-vehicular operations. It is form fitting and in contact with the skin, is comprised of a pattern of flexible, liquid transport ducts woven into the garment fabric, and will maintain the crewman between the limits of sweating and minimum comfort levels for pre-determined work load conditions. It is operational from the PLSS cooling loop only and includes provisions for attachment of biomedical and environmental sensors, and for attachment of a waste management system.

12.3.4 External Thermal Garment (ETG)

The ETG provides thermal insulation for the SSA while the crewman is on the lunar surface. Excluding the ETG gloves, it is worn over the PLSS and the PGA. It is a loose-fitting, multi-layered, two-piece, hooded, pajama type covering with outer surface thermally reflective to solar radiation. A design objective is to enable the crewman to self-don the ETG within LEM.

12.3.4.1 Extra-Vehicular Gloves: Each astronaut is furnished with a pair of extra-vehicular gloves to be worn over the PGA gloves. The EV gloves provide thermal protection and are worn only during lunar exploration. They may be removed for short intervals for added dexterity in performing emergency and

maintenance tasks, or to manipulate and erect mission task equipment. Since the gloves are designed to be taken off at will, a tether is provided linking the gloves to the ETG sleeves.

12. 3. 4. 2 Extra-Vehicular Boots: A pair of extra-vehicular boots is provided each crew member for wear on the lunar surface. They are worn over the PGA boots and, like the thermal garment, have a reflective outer surface. A design objective is to enable the crewmen to self-don the EV boots within LEM.

12. 3. 5 Meteoroid Protection Garment (MPG)

The MPG is a metalized, flexible over-garment worn over the PGA. It protects the PGA and crewman from meteoroid impact during extra-vehicular operation. Studies are in process to determine whether the MPG will be separate or integral with the ETG.

12. 3. 6 Emergency Oxygen System (EOS)

The EOS is a self-contained system containing a limited-time supply of O_2 for emergency PGA pressurization and life support, and is normally activated as a backup for a failed PLSS. It may be coupled to the PGA helmet or removed and stowed. Since it is not rechargeable during the lunar mission, it is to be discarded when expended.